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THESIS

SPACE-BASED SOLAR POWER SYSTEM ARCHITECTURE

by

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December 2012

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13. ABSTRACT (maximum 200 words)

Fossil fuels are, by their very nature, finite resources. There are, however, numerous renewable energy sources that should be taken advantage of. One of the most abundant is also the most difficult to produce on Earth—solar energy.

This thesis explores the feasibility of a space-based solar power satellite. The thesis focuses specifically on the satellite design as opposed to the end-to-end design to include the ground segment. It explores the potential orbits for such a satellite to operate from and ultimately concludes that a geostationary orbit is the only logical location for an operational orbit.

This thesis also focuses on two segments of the spacecraft: the solar array and the power transmission payload. The solar array area was calculated using the current best theoretical solar cells and assumed a 1 GW transmission power. Finally, this thesis explored which transmission payload to recommend for an operational system, concluding that a laser system is the most efficient use of space and weight.

The final portion of this thesis was to examine the business case. Based on the design in this thesis, space-based solar power cannot compete with fossil fuels and likely will not for the foreseeable future.

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SPACE-BASED SOLAR POWER SYSTEM ARCHITECTURE

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LIST OF ACRONYMS AND ABBREVIATIONS

DOC— Department of Commerce

GEO— Geosynchronous Orbit

HEO— Highly Elliptical Orbit

kWh— Kilowatt hours

LEO— Low Earth Orbit

MEO— Medium Earth Orbit

NSSO— National Security Space Office

NASA— National Aeronautics and Space Administration

SBSP— Space Based Solar Power

UCS— Union of Concerned Scientists

ULA— United Launch Alliance

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I. INTRODUCTION

A. BACKGROUND

On July 15, 2010, following the Deepwater Horizon disaster, President of the United States Barack Obama addressed the United States:

> For decades, we have known the days of cheap and easily accessible oil were numbered. For decades, we've talked and talked about the need to end America's century-long addiction to fossil fuels. And for decades, we have failed to act with the sense of urgency that this challenge requires. Time and again, the path forward has been blocked -- not only by oil industry lobbyists, but also by a lack of political courage and candor…We cannot consign our children to this future. The tragedy unfolding on our coast is the most painful and powerful reminder yet that the time to embrace a clean energy future is now. Now is the moment for this generation to embark on a national mission to unleash America's innovation and seize control of our own destiny. (The Office of the President of the United States, 2010).

The April 20, 2010, explosion of the Deepwater Horizon drilling rig in the Gulf of Mexico caused the largest environmental disaster in United States history. These words, spoken by President Obama, demonstrates the heavy reliance the United States has on fossil fuels and the desperate need to move away from using fossil fuels as the main source of energy. There are many alternatives to fossil fuels: nuclear fission reactors, hydroelectric power, wind turbines and solar power to name just a few. Each has advantages and disadvantages but for now, fossil fuels still provide the majority of the world's power.

According to the United States Department of Energy, there is a heavy reliance on non-renewable energy to create electricity. According to the U.S. Department of Energy as shown in Table 1, about 20 percent of all consumed electricity is from renewable sources. Generation of the other 80 percent must then come from non-renewable sources such as petroleum, natural gas, coal, and nuclear. Of the renewable sources that comprise the 20 percent, the combination of solar, tide, and wave sources make up only three-hundredths of a percent of the consumed electricity in 2007.

Electricity (kWh)	
Generation (Annual 2007)	18,778,669,000,000.00
Consumption (Annual 2009)	17,109,665,000,000.00
Total Renewable Electricity Generation (Annual 2007)	3,472,703,000,000.00
Total Solar, Tide and Wave Electricity Generation (Annual 2007)	5,037,000,000.00
Total Renewable Electricity Consumption (Annual 2007)	3,472,703,000,000.00
Percent of Consumed Electricity from Renewable Sources	20.30%
Percent of Consumed Electricity from Solar, Tide and Wave	0.03%

Table 1. Electricity Generation Statistics (After *International Energy Statistics*, 2010)

This table shows that the world must make significant advances before renewable energy sources truly pick up the brunt of electricity generation. Unfortunately, based on the numbers contained in Table 2, one of the prime sources of energy production, coal, is not in infinite supply. Based on the total known reserves in 2005 and the annual consumption rate as of 2008, the world's coal reserves will run out in approximately 2138, leaving 128 years of electricity production before the known reserves are completely gone. This calculation assumes that no new reserves will be found and that the rate of consumption will remain the same as in 2008. It is likely that more reserves will be found but it is also likely that the rate of consumption will increase to serve the ever-increasing demand for electricity.

Coal (short tons)	
Reserves (Total as of 2005)	930,423,000,000.00
Consumption (Annual 2008)	7,238,208,000.00
Approximate length of reserves (in years)	128.54
Approximate year reserves will run out	2138
Years Remaining	128

Table 2. Coal Reserves and Consumption (After *International Energy Statistics*, 2010)

There is another push from governments around the world, and especially in the United States, to decrease the dependence on oil and other petroleum products. As Table 3 shows, the known oil reserves, based on the 2008 rate of consumption, will run out in

approximately 42 years. The problem with this push is that it will put a greater strain on the electrical infrastructure and likely increase the rate of electricity production and therefore coal consumption. Something must be done soon to increase the amount of renewable energy being pushed into the electrical grid soon to meet the rising demand. One possible solution is a space-based solar power infrastructure that provides energy to ground stations strategically placed near high-usage cities across the world.

Petroleum (barrels)	
Reserves (Total as of 2009)	1,342,207,000,000.00
Consumption (Annual 2008)	31,299,365,025.00
Approximate length of reserves	42.88
Approximate year reserves will run out	2052
Years Remaining	42

Table 3. Petroleum Reserves and Consumption (After *International Energy Statistics*, 2010)

One significant disadvantage of the current terrestrial-based solar power systems is the dissipation of the sun's energy as it travels through Earth's atmosphere. Outside of Earth's atmosphere, the solar radiation level is 1367 watts per square meter (W/m^2). However, as stated in the 2007 National Security Space Study on Space Based Solar Power, "by the time it reaches the ground, it has been reduced by atmospheric absorption and scattering; weather; and summer, winter, and day-night cycles to less than an average of 250 W/m^2"; a drop of 1117 W/m^2. Current terrestrial-based solar panels can, at a maximum, absorb only 250 W/m^2. However, if those same solar panels were placed into Earth orbit, those same panels would see an increase of almost 550 percent in the available solar power. The purpose of this thesis is to explore the system architecture required for a space-based solar power system. It will focus specifically on the satellite itself, the energy collection and transfer systems and the orbit.

The main thrust of this thesis, besides the design itself, is to determine whether or not it is cost effective to develop a space-based solar power collection capability. First, this thesis will establish the power requirements based on current fossil fuel-based power

production. Once those power requirements have been established, the satellite system will be designed. Then, based on that design and the established orbit, the launch vehicle can be established. That decision will lead to a cost-to-orbit figure. Using that figure, the estimated satellite cost and an assumed cost of the ground architecture, a comparison can be drawn between the space-based system and traditional fossil fuel-based systems. Based on that comparison, this thesis will recommend areas for further study as well as recommendations on where new technology can make space-based solar power even more cost-efficient.

B. RESEARCH QUESTIONS

In order to guide the research and discussion contained within this thesis, three research questions were proposed. These questions will facilitate not only the design of the satellite architecture, but also whether or not the architecture meets its power objectives. The three research questions are contained in Table 4.

1. What orbit maximizes exposure to sunlight while minimizing cost-to-orbit?
2. Which energy transfer system allows for the most efficient energy transfer?
3. Is a space-based solar power collection system more cost-effective than current energy production methods?

Table 4. Thesis Research Questions

The first question to be answered is what orbit maximizes exposure to sunlight while minimizing cost-to-orbit. A low-earth orbit will not be constantly illuminated by the sun but will be cheaper to get into orbit. A geosynchronous orbit will always be exposed to sunlight but it also costs a lot of money to get a large satellite into that orbit. These two scenarios represent the two extremes when dealing with the orbits. There are two other orbital regimes that might work for this architecture; a middle-earth orbit with a period of approximately twelve hours, and a Molniya orbit which is highly elliptical and has an extended dwell time over the northern hemisphere. Section III will examine the specifics about these four orbits and ultimately make a recommendation on which orbit

makes the most sense. The spacecraft bus will be designed based on the orbital regime it will be operating in.

The second research question is which energy transfer system allows for the most efficient energy transfer. The research for this thesis showed that all of the proposed systems used two methods of energy transfer: microwaves and lasers. Each system has advantages and disadvantages, some of which are based on the operational orbit and some of which are based on the spacecraft and ground system design. Section IV will examine this specific question in-depth and make a recommendation based on that examination.

The final research question is whether or not a space-based solar power collection system is more cost effective than current power production methods. Ultimately, any space-based solar power system will be evaluated based on the dollar per kilowatt-hour metric that current terrestrial-based systems are evaluated on. A baseline system cost will be produced from the examination of the first two research questions as well as the design of the spacecraft and the selection of the launch vehicle. A system-level dollar per kilowatt-hour value, based on that system cost and the estimated energy production, can be calculated. From that, it is fairly easy to draw a comparison between the orbital system and current terrestrial systems.

Answering these three research questions will assist in guiding the discussion on this topic and will also ensure that the main issues surrounding this system are addressed. There are many potential pitfalls and problems with this system so it is important that they are all addressed in one manner or another. Due to the limited scope of this thesis, some of the pitfalls and problems will have to be addressed by making assumptions. This does not mean these issues are not important; they just fall outside the scope of this individual thesis.

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II. THESIS ASSUMPTIONS

A. INTRODUCTION

To be an effective examination of the satellite system, this thesis needs to be focused specifically on the vehicle, the payload and the orbital regime. To design the ground segment and solve the environmental and political issues associated with this system would decrease the amount of time spent examining the critical components of the system and would give this thesis no clear direction but rather would be a general system discussion.

To facilitate the proposed direction of this thesis, this section will establish assumptions from which the remainder of the thesis will draw. There are three groups of assumptions that will be addressed in this section. The first is general assumptions about the ground segment; the number of ground stations, their locations and a description of the general system. The second group is about the impact the system will have on the environment. The final group is dedicated to discussing the political issues involved with this system. These three groups of assumptions will provide the framework from which the rest of the system will be designed.

These assumptions are meant to be neither an all-inclusive, exhaustive list nor highly and thoroughly detailed. Rather, these assumptions identify some of the potential problems and pitfalls of creating a space-based solar power system, and perhaps more importantly to this thesis, the issues that need to be solved before answering the critical questions posed by this thesis. Solutions to these problems have proposed in order to facilitate the remainder of the thesis, but these solutions are not perfect and would likely need more discussion and detail in order to continue the system discussion beyond this thesis.

B. GROUND SEGMENT ASSUMPTIONS

The first set of assumptions for this thesis deal with the ground system segment. This thesis will focus on the space vehicle, the energy transfer subsystem and the vehicle orbit. The discussion on the ground segment, while critical to the overall system, will not

be discussed in detail in this thesis. Instead, the remainder of this thesis will design the vehicle based upon the ground segment as described in this section. This identifies the importance of the ground segment while giving space and time to the discussion on vehicle architecture.

Prior to any discussion about number and placement of ground stations, the proposed ground stations must have an estimated footprint. The larger the ground station, the more constrained the placement will be. One issue with making an assumption about the size of the ground segment before any other discussion about the system architecture is that one major component has not yet been discussed, let alone selected. The method of energy transfer plays a large role in the size of the ground station because it directly impacts the size of the receiver. The two proposed energy transfer methods, laser and microwave, have very different ground footprints. Also impacting the size of the footprint is the orbit from which the vehicle is operating. As is fairly obvious, if the vehicle is farther away, the energy is going to spread out more before reaching the receiver. This is true regardless of the method of transmission.

For the purpose of this thesis, due to the unknowns as presented thus far and due to the potential risks associated with transferring energy as described in some of the following sections, the ground station footprint will be a ten kilometer square (NSSO, 2007, p. 7). This assumption will allow for a safety buffer regardless of which transmission method is chosen and will allow the thesis to move forward to selecting the number and location of the ground stations.

The ground stations from which the vehicle will operate must be optimally placed based upon the orbital regime in which the vehicle is operating as well as the contact time required to command the vehicle. In this case, a significant issue with the ground segment will be proving the technology works while operating with minimum investment to ensure that the system is as viable as it appears on paper. This is a significant problem because it potentially impacts the number of ground stations that would be built to receive power sent from the satellite to Earth. Depending on the orbital regime the vehicle is operating in, this could adversely impact the perceived viability of the project.

If the selected orbit is a geosynchronous orbit (GEO), a single ground station would suffice for vehicle checkout and demonstration. However, if the vehicle is in a low-earth orbit (LEO), a single ground station would not be sufficient to demonstrate the economic viability of the project. A vehicle in LEO would only be in view of a single ground station for approximately 9–15 minutes a few times a day while a vehicle operating at GEO would have constant access to a ground station and would be illuminated by the sun for the majority of each day. Since the final answer to this issue depends on the orbital regime selected in section III, this section will make a few assumptions about the number and placement of the ground stations.

For the purposes of this thesis, if the vehicle is operating in a geosynchronous orbit, only one ground station is required, but more should be built within view of the final orbit. Multiple ground stations within view of the vehicle would provide options for power distribution in case of mechanical problems or bad weather at other sites. Due to the large footprint of the ground station as suggested above and the unique power requirements of some locales, the following general locations have been selected for ground stations: California, New York, Brazil, Hawaii and Thule Air Base, Greenland.

During this thesis, references will be made to a system based around a vehicle operating in a low-earth orbit. Due to the fact that a LEO vehicle will only be in contact with a single ground station for 9–15 minutes (difference depending on altitude and ground track), multiple ground stations must be used. In this configuration, based on even spacing for constant contact with the vehicle, the following locations will be used for ground stations: California, New York, Brazil, Hawaii, Thule Air Base, Greenland, England, Moscow, Johannesburg, Diego Garcia, Tokyo and Australia. The proposed ground stations are shown in Figure 1.

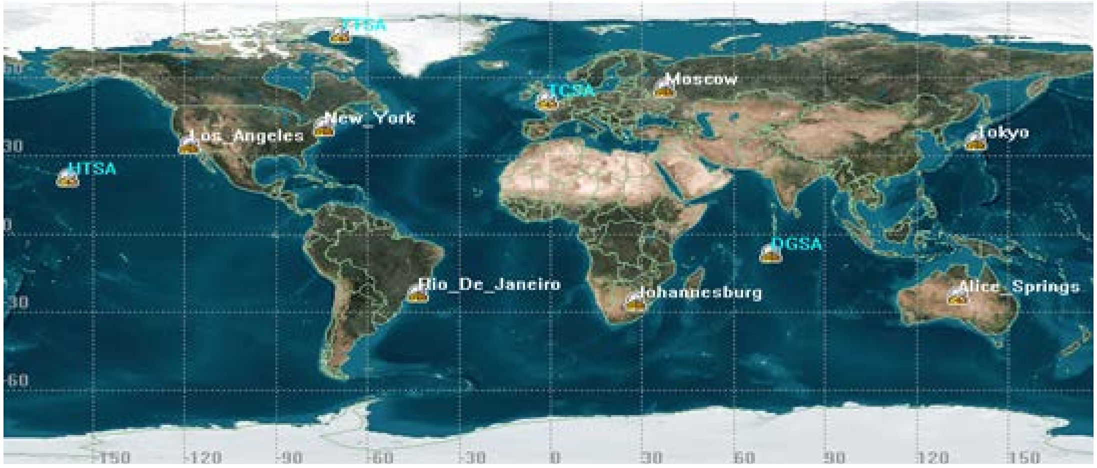


Figure 1. Proposed Ground Stations

C. ENVIRONMENTAL IMPACT ASSUMPTIONS

Any proposed space-based solar power system is not without potential environmental problems. One issue is the sheer size of the ground segment. As stated previously, a microwave receiving antenna, or rectenna, can be on the order of 10 kilometers in diameter. That by itself can have a major impact on the surrounding environment. However, as stated in the 2007 Interim report by the National Security Space Office,

> microwave receiving rectennas allow greater than 90 percent of ambient light to pass through, but absorb almost all of the beamed energy, generating less waste heat than terrestrial solar systems because of greater coupling efficiency. This means that the area underneath the rectenna can continue to be used for agricultural or pastoral purposes. (p. 29).

Another possible environmental impact would be the high level of energy being transmitted from space to the Earth's surface. As stated above, a microwave rectenna absorbs almost all of the transmitted energy, so there should not be any problems related to microwave energy on the Earth's surface. However, if a laser system were used, there could be other serious problems related to the ground segment. Most importantly, there could be no error in pointing, unless a large margin of error was built into the ground segment. An extremely high-powered laser could give people or wildlife serious burns or cause blindness and possibly even death depending on exposure proximity and time.

For the purposes of this thesis, the environmental impacts such as stated above are negated by the large footprint of the ground station as well as proper education of the surrounding populous. The microwave rectenna appears, based on presented research, to be up to 10 km in diameter. If the ground station footprint is a 10 km square, there is always a buffer between the edge of the rectenna and the edge of the fenced area. That area provides the buffer for any potential pointing inaccuracies. A laser system has a significantly smaller footprint so the 10 kilometer square ground station footprint will have no problem providing a significant buffer to account for any inaccuracies.

In the NSSO report (2007), it states that “the SBSP Study Group found that when people are first introduced to this subject, the key expressed concerns are centered around safety, possible weaponization of the beam, and vulnerability of the satellite, all of which must be addressed with education” (p. 26). Proper education of the surrounding populous will demonstrate the risks and benefits of a space based solar power system and will establish that the inherent risks (regardless of transmission technology) are well worth the capability that a space based solar power satellite will provide.

D. POLITICAL RELATIONSHIP ASSUMPTIONS

Regardless of the final system architecture that is selected, numerous political relationships must be forged and maintained in order for the system to be successful. As stated in the ground station assumptions section, there will be ground stations in many countries, no matter what configuration the space segment is in. The only difference between a geosynchronous and low-earth orbit system is the number of countries that would need to “buy in” to the system. In the geosynchronous system, the initial focus is on the western hemisphere in high-density population centers or places that have unique power requirements. The countries involved in the GEO system are the United States, Brazil and Greenland. The LEO system involves much more international cooperation, since there are proposed ground stations in the United States, Brazil, Greenland, England, Russia, South Africa, Japan and Australia.

International partnerships are always wrought with potential problems. Any time multiple governments have to work together towards one goal, it can be very difficult.

Either system, whether based in LEO or GEO, will foster competition for the system resources. It will be extremely difficult for all of the countries to agree on how and when to use the payload and how to share the time based on need, availability and weather. Sometimes, these issues can be worked out based on monetary investment, technical assistance, preponderance of assets (ground stations, vehicles, etc.). Most often, these solutions lead only to more problems in the partnership. For the purpose of this thesis, assume that this partnership has been ironed out completely and there is no competition for assets. For the GEO based system, the default user will be the United States, based on investment, preponderance of assets and a much higher power requirement. When the United States cannot use the asset due to weather, maintenance or other issues preventing its use, the asset will be transferred to Brazil for use. A LEO-based system will have to constantly share the asset or assets because no one country can always be in contact with the vehicle all the time. This can be done through simple scheduling and averaging the time each ground station has in contact with the vehicle.

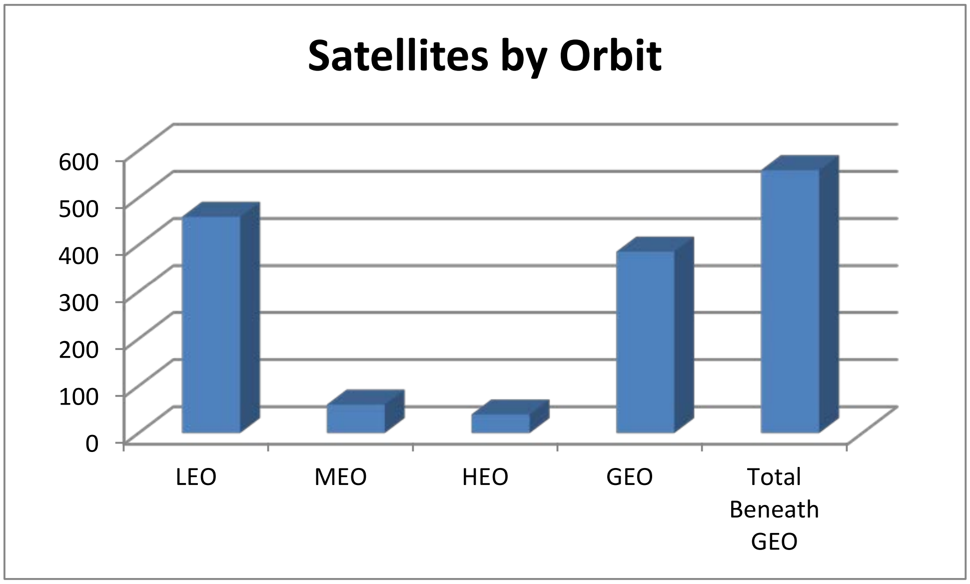


Figure 2. Operational Satellites by Orbit Type (From Union of Concerned Scientists, 2010)

Another potential problem that will have to be dealt with on an international scale is the potential to radiate other satellites with either a high-power microwave emitter or a high-energy laser. Either one can have catastrophic consequences if they encounter another satellite. According to a database published by the Union of Concerned Scientists (UCS), approximately 60 percent of operational satellites are in orbits other than a geosynchronous orbit. This places 558 of 943 satellites in a hazardous orbital regime with respect to a space-based solar power satellite operating in a geosynchronous orbit as shown in Table 5 and Figure 2. Based on this data, a vehicle operating in a low-earth orbit will encounter this conjunction problem less frequently than a vehicle operating in a geosynchronous orbit since the majority of satellites (60 percent) will be between a GEO vehicle and the Earth. The easiest way to mitigate the problem of accidental radiation is to deconflict between the vehicle and the rest of the satellite catalog as maintained by the Joint Space Operations Center (JSpOC). Based upon the orbit of the vehicle, the ground station accesses and the orbits of other satellites as provided by the JSpOC, the solar power vehicle should be able to avoid radiating other satellites.

Orbit	Number	Percentage
LEO	459	48.67%
MEO	60	6.36%
HEO	39	4.14%
GEO	385	40.83%
Total Beneath GEO	558	59.17%
Total	943	

Table 5. Operational Satellites by Orbit Type (After Union of Concerned Scientists, 2010)

Besides the issue of accidental radiation, it is also significantly sensitive because of the potential for the vehicle to be used as a weapon. Both a microwave transmitter and a laser, as stated previously, would make a great weapon that could be trained on other satellites. For this reason, the vehicle should not be controlled by one government specifically. Preferably, it would be operated by a corporate civilian entity. If it becomes necessary for a government to operate this system, it should be operated by a coalition

with representation from multiple governments to ensure that the vehicle is not being misused. The oversight as provided by multiple governments should allow for enough transparency so as to avoid any perception of wrongdoing, even if an accidental radiation event should occur.

E. LAUNCH COST ASSUMPTIONS

In order to facilitate the discussion on orbital regimes, a general assumption must be made about the cost to launch satellites into different orbits. Cost assumptions are always problematic and potentially plagued with budgetary and schedule overruns which inflate the launch cost. For the purpose of this thesis, any discussion on launch cost revolves strictly around the dollar-per-kilogram metric and does not factor in any other numbers besides the vehicle payload and the dollar figure of the launch vehicle.

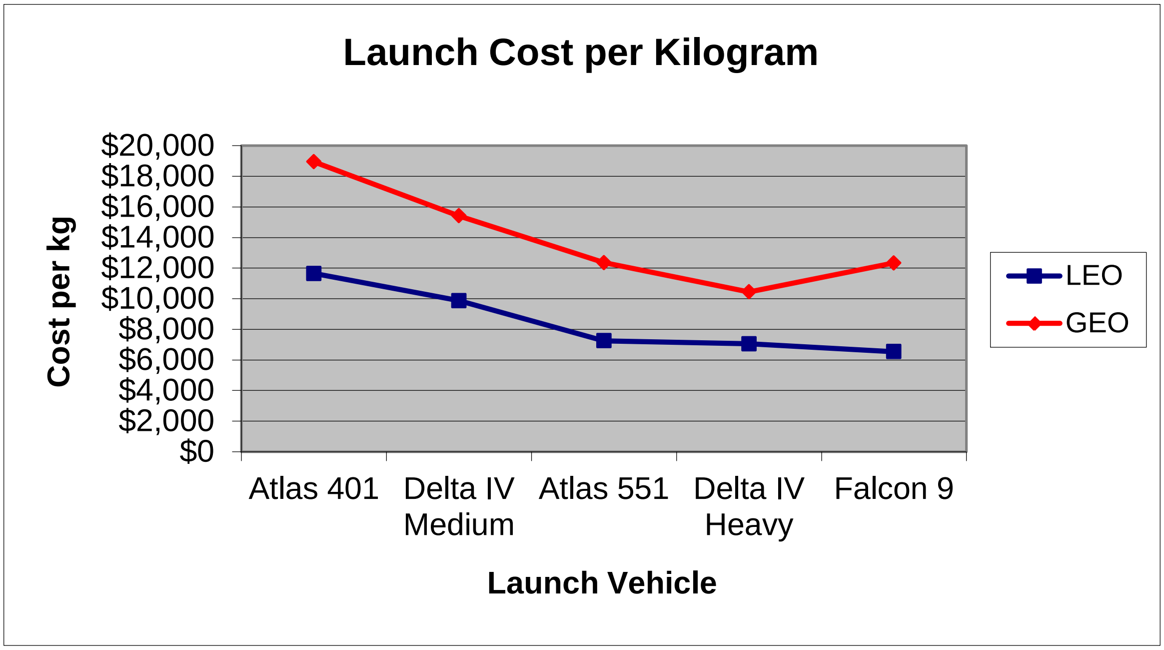


Figure 3. Launch Cost Per Kilogram of Maximum Payload Weight

By using this single metric, a general assumption can be made about launching satellites into orbit; it is more expensive to launch a satellite into a geosynchronous orbit than it is to launch into a low-earth orbit. For these cost discussions, the single dollar-per-

kilogram metric was chosen because it limits the variables being discussed to the cost of the launch vehicle itself and the payload that it can carry. Using the handbooks provided by the United Launch Alliance (ULA) and Space Exploration Technologies Corporation (SpaceX), five launch vehicles capable of sending large payloads to both LEO and GEO were selected. These five vehicles represent the heavy-lift capability of the United States. The payloads of these five vehicles were then compared to their payload weights to both LEO and GEO and those dollar-per-kilogram figures plotted together for trend analysis. When plotted, a trend emerges: it is, in fact, cheaper to launch into LEO than it is into GEO.

Vehicle	LEO (kg)	GTO (kg)	Cost ($M)	Cost per kg to LEO	Cost Per kg to GEO
Atlas 401	7724	4750	$90,000,000	$11,652	$18,947
Delta IV Medium	7087	4541	$70,000,000	$9,877	$15,415
Atlas 551	15179	8900	$110,000,000	$7,247	$12,360
Delta IV Heavy	19839	13399	$140,000,000	$7,057	$10,449
Falcon 9	8560	4540	$56,000,000	$6,542	$12,335

Table 6. Launch Cost Per Kilogram of Maximum Payload Weight (From United Launch Alliance, 2010a and 2010b and Space Exploration Technologies Corporation, 2009)

This launch cost analysis did not end with a single launch vehicle cost. There are other potential problems that arise when launching large payloads, not the least of which is the payload fairing size. In this launch vehicle class, the payload fairings are four to five meters. While this is not an insignificant size, it potentially pales in comparison to microwave rectennas and large solar panels. Multiple launch vehicles may be required to place all the pieces into orbit, thus driving up costs. To illustrate this potential problem, a vehicle weight was chosen that exceeded the maximum payload weight of all five vehicles. At this point, an assumption was made that the vehicle could be broken into parts to roughly fill the vehicle to its maximum capacity where necessary. From that, the number of launch vehicles needed to launch the payload weight was generated, compared to the launch vehicle cost, and the dollar-per-kilogram metric was again produced. While the calculations and assumptions are clearly not perfect, since a satellite, in reality, cannot

arbitrarily be broken into pieces, this analysis offers an order of magnitude and a general trend from which to draw conclusions.

The second evaluation of the launch cost shows that, with increasing payload size, the dollar-per-kilogram launch cost will increase significantly, especially if the vehicle does not or cannot fill an entire launch vehicle. Leaving empty space inside a payload fairing of a launch vehicle will always drive an increase in launch cost since the cost of the launch vehicle is spread out over a smaller payload size. It is therefore imperative that during the design of the vehicle, significant thought is placed on the vehicle weight as a whole and also on the size of specific portions of the vehicle that could be broken apart to fit into multiple launch vehicles to be assembled on orbit. By planning for these issues during the design phase, launch costs can be lowered and schedule and budget overruns can be avoided.

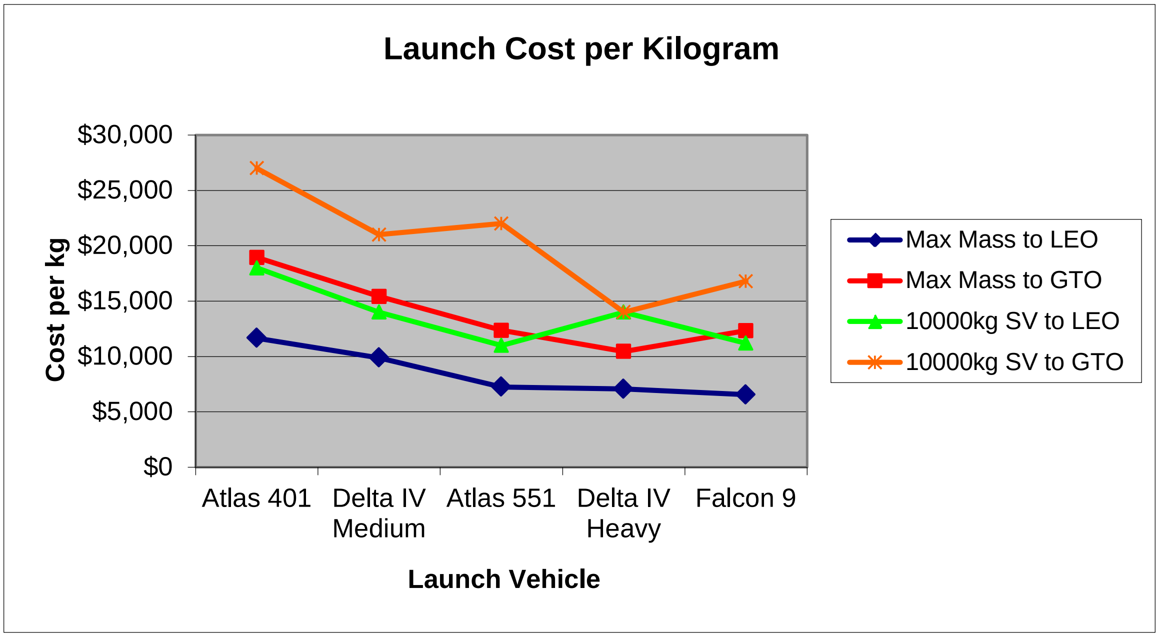


Figure 4. Launch Cost Per Kilogram for Multiple Launches

Vehicle	Weight	No LVs to LEO	No LVs to GEO	Cost to LEO	Cost to GEO	LEO (20000kg SV)	Cost per kg to GEO (120000kg SV)
Atlas 401	20000	3	5	$270,000,000	$450,000,000	$13,500	$22,500
Delta IV Medium	20000	3	5	$210,000,000	$350,000,000	$10,500	$17,500
Atlas 551	20000	2	3	$220,000,000	$330,000,000	$11,000	$16,500
Delta IV Heavy	20000	2	2	$280,000,000	$280,000,000	$14,000	$14,000
Falcon 9	20000	3	5	$168,000,000	$280,000,000	$8,400	$14,000

Table 7. Launch Cost Per Kilogram of 20000kg Payload (After United Launch Alliance, 2010a and 2010b and Space Exploration Technologies Corporation, 2009)

While this analysis could appear to be obvious or superfluous, it is critical to establish significant cost increases or savings due to the overall architecture metric: the dollar-per-kilowatt. An increased launch cost could significantly increase the dollar-per-kilowatt figure out of the usable range. By analyzing the cost-to-orbit figure and establish an assumption, it becomes easier to make decisions such as those made in the coming chapters. The NSSO also realized the importance of cheap, reliable access to space as a key enabler of the space based solar power architecture. In the 2007 report, the NSSO Study Group states that "SBSP cannot be constructed without safe, frequent (daily/weekly), cheap, and reliable access to space and ubiquitous in-space operations. The sheer volume and number of flights into space, and the efficiencies reached by those high volumes is game-changing. By lowering the cost to orbit so substantially, and by providing safe and routine access, entirely new industries and possibilities open up" (p. 12).

F. CHAPTER SUMMARY

The issues discussed in this chapter are by no means the exhaustive list of the potential problems associated with this system. It is also not a complete discussion on the issues presented. This chapter is meant as a way to establish a framework from which to build the rest of the system architecture. Since this thesis is focusing strictly on the vehicle, many of the issues presented in this chapter have nothing to do with the vehicle directly. Instead, this chapter focused on the ground segment, the potential environmental issues and the political influences.

The ground segment discussion focused on the two types of proposed energy transfer systems, microwave and laser, and their impact on the ground segment footprint. Between the two systems, the microwave rectenna had the greatest footprint so the assumed size of the ground station was designed around that. Then, based on energy usage levels, available space and other factors, candidate ground stations were identified. The specific location of the ground stations are dependent on which orbit the satellite is operating in (geosynchronous versus low-earth orbit).

The next issues discussed in this chapter were environment-related. The first issue was the potential problem stemming from the huge microwave rectenna, but based on the construction of the rectenna, there should not be any environmental impact. Also, based on the sheer amount of energy being transmitted to the surface, there could always be environmental and health-related problems. Most of the environment-related issues were mitigated, at least locally, by the size of the ground station which was built with sufficient margin to include a buffer to the surrounding environment.

The final issues discussed in this chapter were political. Working with international partners in a project this size could generate numerous problems. Another identified problem was radiating satellites between the vehicle and the Earth with high-power microwaves or lasers. Proper planning and prior agreements between international partners prevents many of the issues that were identified in this chapter. For the purpose of this thesis, these issues are already solved based on the factors discussed.

The assumptions presented in this chapter are an important step towards designing a space-based solar power system. Without these assumptions, the remainder of this thesis could not be presented in the manner that it is. The assumptions provide a strong basis from which to build the satellite and the orbit and, from there, identify whether or not this system is viable based on the amount of energy collected versus the amount of money spent on development, launch and operations. Now that the assumptions have been established, it is time to start designing the vehicle. Vehicle design starts with establishing the operating environment: the orbit.

III. SATELLITE ORBIT

A. INTRODUCTION

With the assumptions established this thesis now turns to establishing the operational orbit. System orbitology is critically important in this discussion. It influences four important design points: sun exposure, location of ground stations, periods of access to those ground stations and the distance over which the energy must be transmitted. The orbit, however, is one of the two critical factors in determining the amount of money spent on getting the vehicle into orbit. Therefore, there is a delicate balance between generating the right amount of power, getting that power to the ground and how much money is spent getting the vehicle into its operational orbit.

For example, a low-earth orbit will cost the least amount of money to get the vehicle into orbit, even if it's quite heavy. However, it's exposure to the sun will be significantly less per orbit than other orbits. It will also require more ground support as its dwell time per site will be between 9 and 15 minutes. A significant upside though is that the transmitted energy has a short distance to travel so the transfer system will not drain as much power. A geosynchronous system on the other hand, has a different set of advantages and disadvantages. It will cost a lot of money to get a heavy vehicle into that orbit. The transferred energy also has a very long distance to travel which increases the power requirements and potential pointing inaccuracies.

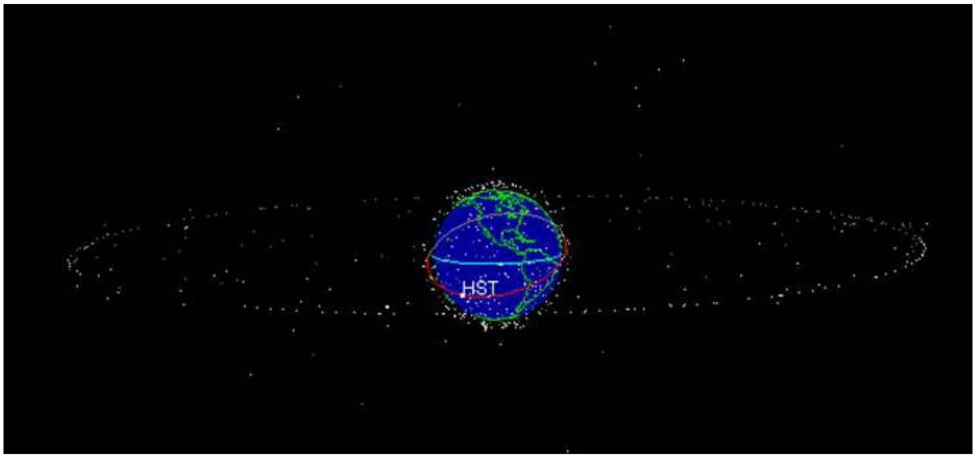


Figure 5. Satellites In Orbit (From National Aeronautics and Space Administration, 2010)

Another potential issue depending on the orbit is the exposure of other satellites to high levels of transmitted energy. As the orbit of the solar power satellite gets higher, more satellites are potentially exposed to the energy it transmits. The great majority of space objects are located in low-earth orbit regimes, as shown in Figure 5, so a vehicle located in a geosynchronous orbit will have a much higher chance of radiating another satellite than a vehicle located in a low-earth orbit.

B. CANDIDATE ORBITS AND THEIR PROPERTIES

There are four general orbits that all satellites fall into: low earth orbit (LEO), middle earth orbit (MEO), highly elliptical orbit (HEO), and geosynchronous orbit (GEO). All four orbits have widely varying attributes, so coming to an ultimate conclusion on which to use operationally can be difficult. This section focuses on the four different orbits and their strengths and weaknesses, along with potential mitigation strategies for the weaknesses.

1. Low Earth Orbit (LEO)

The first orbit under consideration is a low-earth orbit. It is generally accepted that a low-earth orbit extends out to about 1000 kilometers altitude, but most LEO satellites operate beneath that altitude. Since a low-earth orbit has a wide range of operating altitudes, it can be difficult to choose a specific altitude. For this specific

system, the vehicle will operate at an altitude of 800 kilometers. This altitude was chosen specifically for its higher dwell time over a ground station. At 800 km, the vehicle will be in view of a ground station for approximately 15 minutes. The slightly higher altitude also leads to longer sun exposure which is good for this architecture.

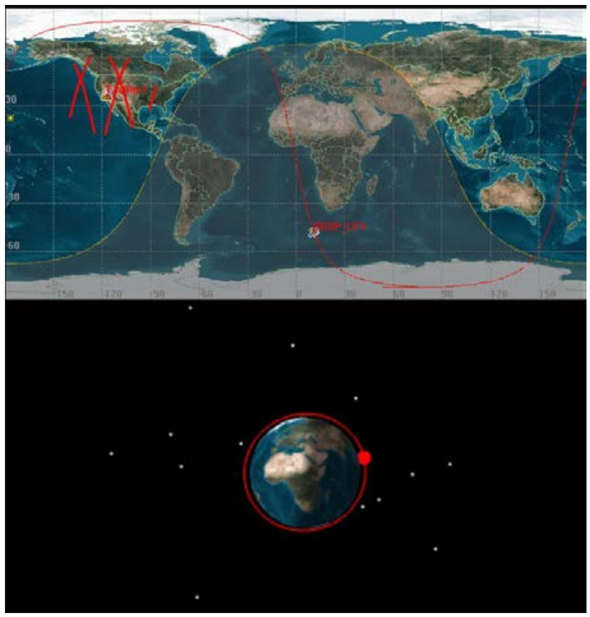


Figure 6. Low Earth Orbit

The scenario was modeled in the Satellite Tool Kit (STK) in two different ways. One scenario was a single ground station and the other scenario was the full set of ground stations as established in the assumptions. This was done because, more than likely, the entire ground segment will not be built at once. It will have to go through a demonstration phase before the complete architecture is built. These two scenarios were run separately for the sake of comparison between the four possible orbits.

Figure 6 shows the vehicle in a low-earth orbit with an altitude of 800 kilometers. It is in a sun-synchronous orbit with an inclination of 98.6º. In this scenario, access was run against a single ground station, located in the Mojave Desert. Due to the unique requirements of the vehicle, the only access times considered valid were when the vehicle was in direct sunlight. If that were not the case, there would be artificial access times in which the vehicle could not transmit energy down to the ground station. Due to the short

dwell time and only having a single ground station, there are only five valid access times. The access times are listed below in Table 8.

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)	Duration (min)
1	7/8/2010 18:13	7/8/2010 18:17	229.987	3.833
2	7/8/2010 19:46	7/8/2010 19:58	762.712	12.712
3	7/8/2010 21:22	7/8/2010 21:32	602.589	10.043
4	7/9/2010 6:35	7/9/2010 6:47	718.998	11.983
5	7/9/2010 8:11	7/9/2010 8:23	699.011	11.650

Table 8. Low Earth Orbit Access Times for Single Ground Station

As Table 8 and Figure 7 both show, these access times are certainly not ideal. The first access is only 3.8 minutes long and the others are in two groups with access lengths between 10 and 13 minutes. This means that in a single 24-hour period, the vehicle can transmit energy back to the ground station for a total of 50.221 minutes; less than an hour. Figure 7 shows graphically the access times over the entire 24-hour period. It is very obvious that there are serious holes in the access availabilities for a low-earth orbit vehicle contacting a single ground station.

The primary problem with this outcome is that a system based in low-earth orbit would likely have to use this configuration in order to prove its viability. It will be difficult to sell other countries on the cost effectiveness of the system if it cannot be proven. However, if the proof-of-concept design uses a single vehicle and a single ground station, it will be almost impossible to prove the economic viability of the system, due to the fact that the vehicle has no illuminated access to the ground station for long periods of time.

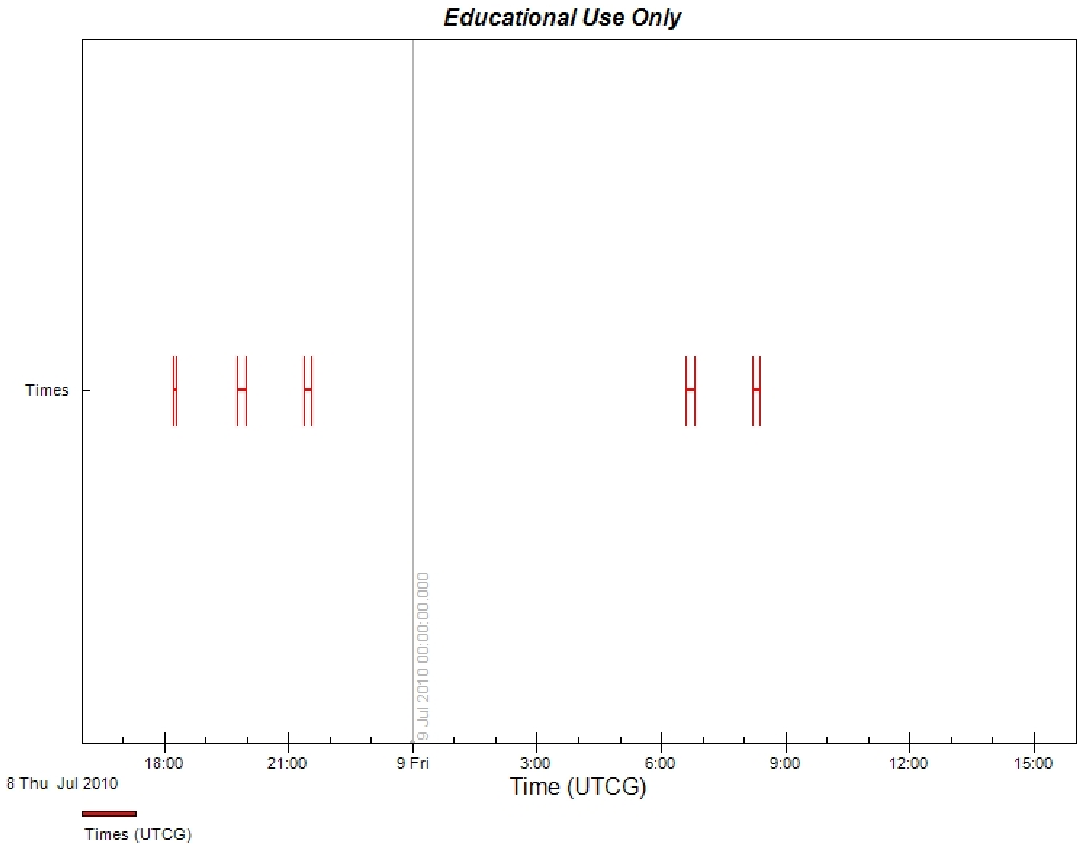


Figure 7. Low Earth Orbit Access Times for Single Ground Station

The alternative, of course, is to prove the economic viability by showing that the access times of the system are significantly better if multiple ground stations are used. To prove this point, the same vehicle in the same orbit is run against the full complement of 11 ground stations. If this is the case, the access times look significantly better than the previous scenario. Figure 8 shows the graphical representation of the access times for a LEO vehicle accessing 11 separate ground stations spread across the world.

The graph may not be conclusive, but the access times as shown in Table 9 are. As stated above, the total access time for a single satellite and for a single ground station was 50.221 minutes, which is less than an hour. The total time as shown in this scenario with 11 ground stations is 681.870 minutes, which comes out to 11.36 hours – almost half a day. That amount of access time is a very significant increase over that provided by a single ground site and makes a huge difference in the economic viability of the system.

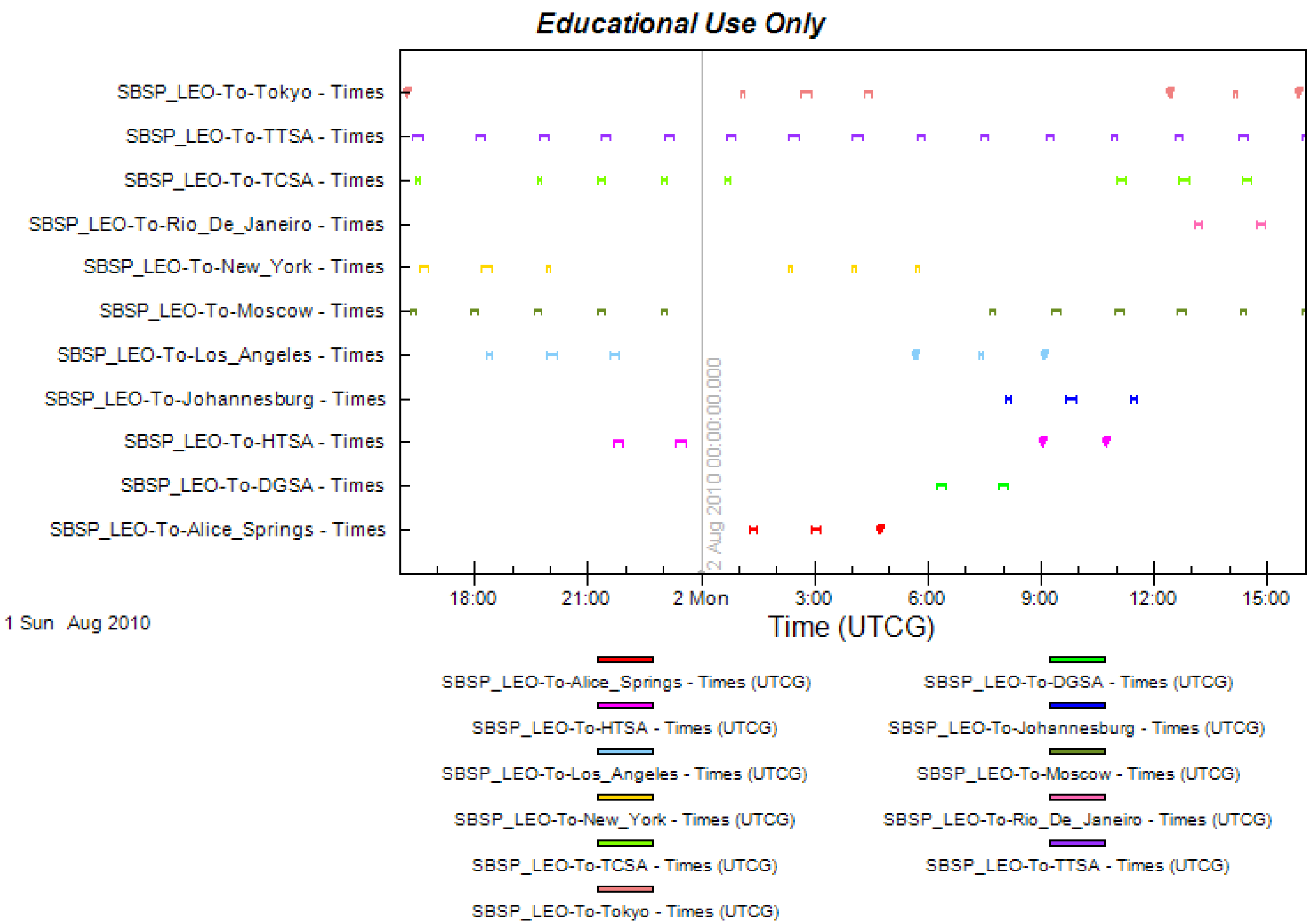


Figure 8. Low Earth Orbit Access Times for Multiple Ground Stations

A low-earth orbit carries many potential advantages to the overall system but comes with unique challenges. A low-earth orbit means that the cost to launch the system will be lower. Part of that equation is the vehicle weight, which will be significantly decreased because the power transmission payload will not have to transmit over a long distance meaning that the payload can cut weight. Another advantage is that, while each individual ground station will not have an extended dwell time, the possibility of a proliferation of ground stations across the globe could provide near-constant transmission, thereby maximizing the potential of the satellite. This would, however, require a heavy investment in the ground stations due to the sheer number. A significant disadvantage to this orbit is the shorter dwell time, which reaches a maximum at about 15 minutes. While multiple accesses to the ground station will help, the short dwell time does affect the overall system because half of the day is still spent not transmitting energy back to the Earth.

Total Access Time	Duration (sec)	Duration (min)
	40912.197	681.870
Average	3719.29	61.99

Table 9. Average LEO Access Times for Eleven Ground Stations

2. Middle Earth Orbit (MEO)

Another possible orbit for this system is a semi-synchronous orbit, or a middle earth orbit (MEO). This orbit is special because it has a period of exactly half a day, placing it at an altitude of 20,200 kilometers. The two main advantages to this orbit are a much longer dwell time over the target than a LEO vehicle and a lower cost-to-orbit figure than a geosynchronous orbit. It will, however, be more expensive than launching into a low-earth orbit and will not have the dwell time of a geosynchronous orbit. It might, however, prove to be a compromise between the two. The designed orbit also has an inclination of 45 degrees, which does establish a significant limit on some of the

ground stations, such as Thule Air Base in Greenland. Rather than being in view for approximately 12 hours at a time, the accesses to Thule AB will be shorter.

The first scenario shown below is much like the previous section, in which a single vehicle is run against a single demonstration ground station. This was again done to show a possible proof-of-concept and viability exercise. Depending on the outcome, it may appear as though the orbit is not viable even though it may be with a full complement of ground stations. The second scenario shows the accesses from the vehicle to the full set of ground stations demonstrating a fully operational capability.

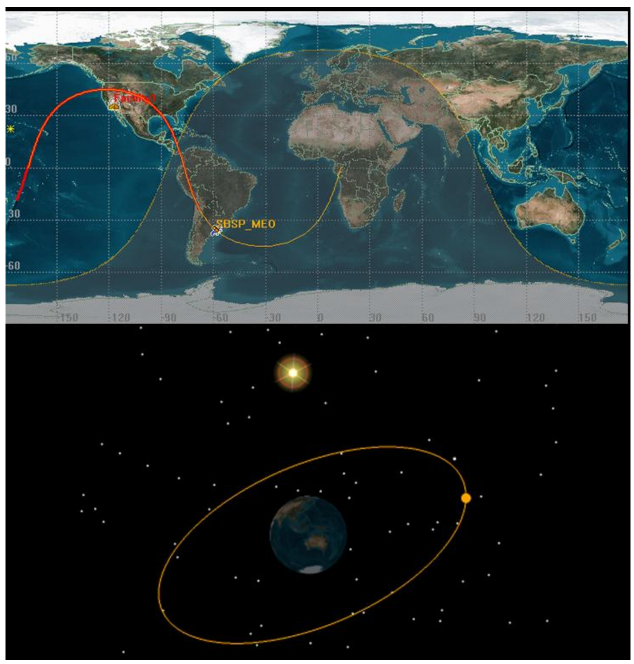


Figure 9. Semi-synchronous or Middle Earth Orbit

Clearly, if the single ground station system is used in a viability exercise, it will appear drastically different than making observations based on the full system with all 11 ground stations. If only one ground station is present, the vehicle experiences significant periods of no contact each day. That period is significantly longer than the multiple contacts experienced by the low-earth orbit vehicle accessing a single ground station. However, if the entire complement of ground stations is used, the vehicle is always in

contact with a ground station, meaning that this scenario is very economical as far as transmitted energy is concerned.

As shown in Tables 10 and 11, the access times are drastically different between the single ground station and the multiple ground station scenarios. The total time in view of the single ground station is 485 minutes, or approximately 8 hours. Taking all 11 ground stations into account, the vehicle is in view of the ground stations for 5921 minutes, or 98 hours. Since this scenario was run for a 24-hour period, the vehicle is in view of multiple ground stations at the same time at some points in its orbit. This is very advantageous for this type of system because it gives the vehicle the option to change ground stations based on weather or maintenance.

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)	Duration (min)
1	7/8/2010 16:00	7/8/2010 23:09	25784.603	429.7433833
2	7/9/2010 15:04	7/9/2010 16:00	3344.492	55.74153333
Total			29129.095	485.4849167

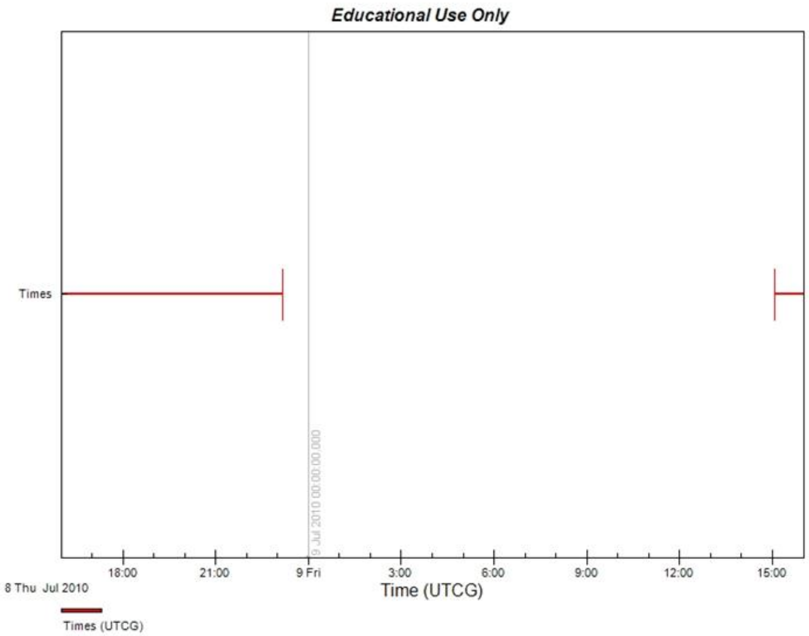
Table 10. MEO Access Times for Single Ground Station

This semi-synchronous orbit presents some distinct advantages as compared to the low-earth orbit. First, the access times are significantly increased, from 11 hours to 98 hours. Along with that, there are almost always multiple ground stations in view of the vehicle at any one time, providing alternatives to ground stations that might be experiencing bad weather or undergoing maintenance. Another advantage of this orbit is that, being at 20,200 kilometers altitude instead of [A1]800, the atmospheric drag on the vehicle is significantly decreased.

Total Access Time	Duration (sec)	Duration (min)
	355260.4	5921.007
Average	32296.4	538.27

Table 11. MEO Access Times with Eleven Ground Stations

At the same time, however, there are disadvantages to this orbit. The increased altitude is going to cost more to get into orbit. Along the same lines, it also means that the vehicle will have to start cutting weight, as a single launch vehicle has a lower weight limit in this orbit as opposed to a low-earth orbit. Another significant downside to this orbit, as discussed in the assumptions section, is that the majority of satellites in orbit are going to be between this vehicle and the ground stations, thereby increasing the vulnerability of being radiated or lased, neither of which is healthy for a satellite.



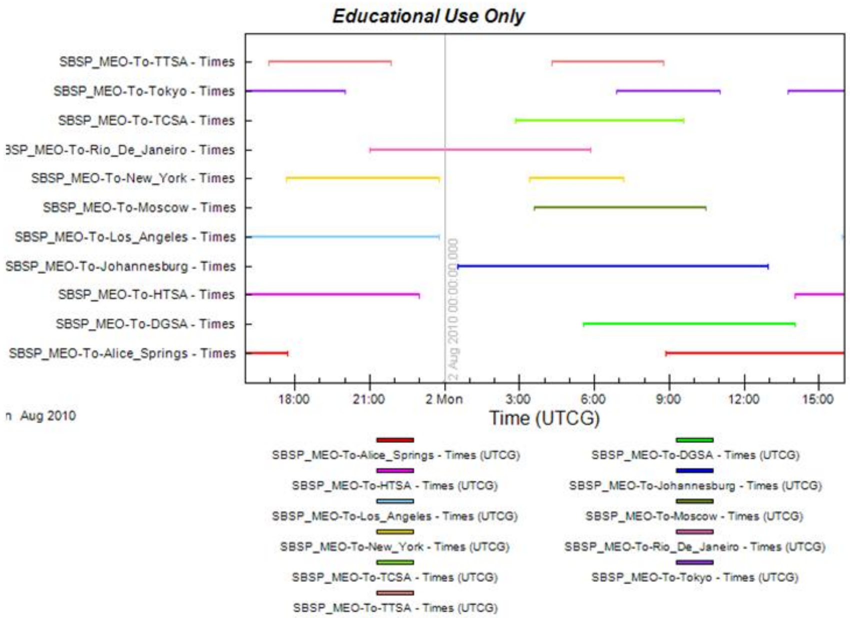


Figure 10. Semi-synchronous or Middle Earth Orbit Access Times

3. Highly Elliptical Orbit (HEO)

The next potential orbit is the highly elliptical orbit (HEO) also known as the Molniya orbit. The purpose of this orbit is to spend the majority of the orbit over a hemisphere of interest. It is very elliptical and highly inclined as well, leading to very interesting accesses and ground tracks. This orbit has an inclination of 63.4 degrees, an eccentricity of 0.74 and a semi-major axis of 26,553 kilometers. In different terms, the orbit altitude at perigee (the closest point to the Earth's surface) is 500 kilometers. The altitude at apogee (the farthest point from the Earth's surface) is 39,850 kilometers. This accounts for the high eccentricity of the orbit. In the example provided, the HEO orbit is focused on the northern hemisphere, but by changing orbital parameters, this orbit could support southern hemisphere users as well.

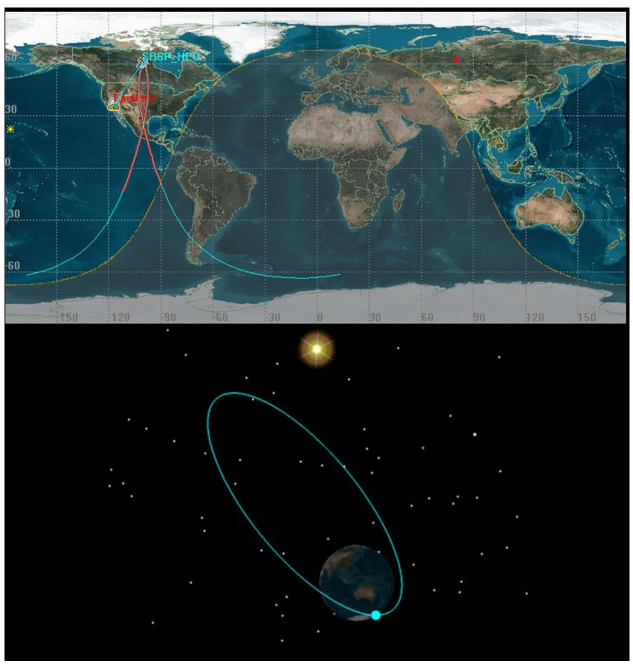


Figure 11. Highly Elliptical Orbit

For this orbit, the same formula was followed for testing different ground configurations. The first configuration was, again, a single ground station in the Mojave Desert, designed to provide insight into what would happen if a proof-of-concept design was required before full operational capability was built. The second configuration was

the full complement of 11 ground stations, the same as the last two examined orbits. The differences between these two results will provide necessary insight into the potential pitfalls of using a limited ground configuration to prove the concept.

Figure 12 shows the access graphs for the two configurations, with the single ground station configuration on the left and the multiple ground station configuration scenario on the right. These graphs will look somewhat different than the last two based on the orbit parameters and where the vehicle is during the accesses. In other words, where the last two orbits have been fairly consistent from one access to the next, these accesses can vary drastically from one orbit to the next based on the latitude of the ground station and the location of the vehicle in the orbit.

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)	Duration (min)
1	7/8/2010 16:19	7/9/2010 3:36	40610.456	676.8409333
2	7/9/2010 8:15	7/9/2010 11:35	11965.998	199.4333
Total				876.2742333

Table 12. HEO Access Times for Single Ground Station

Clearly, this orbit has again increased the access time over the last two, at least for a portion of the 24-hour simulation period. These accesses show that the ground stations are visible to the vehicle for much longer periods of time with a shorter period of no contact in between. This is a very promising scenario for this vehicle, since extended contact with a ground station improves the economic viability of the system. Tables 12 and 13 show this data in table form with the total access time associated with both scenarios.

Total Access Time	Duration (sec)	Duration (min)
	476337.113	7938.952
Average	43303.374	721.723

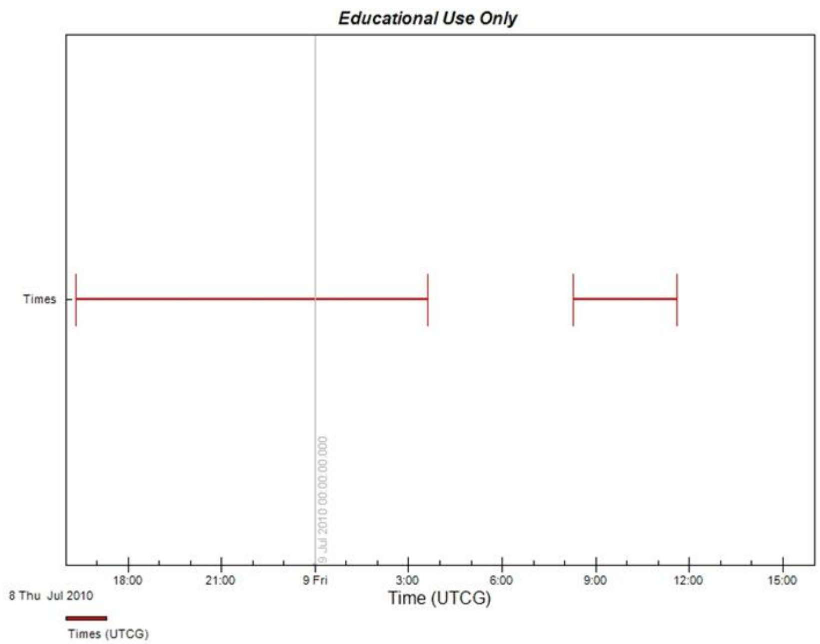
Table 13. HEO Access Times for Eleven Ground Stations

These tables tell a very similar story to the access graphs depicted in Figure 12. While the overall number of accesses has decreased significantly, the time in access has again increased over the last examined orbit. The total time in access of the single ground station is 876 minutes, or 14.6 hours. When this orbit is examined against all 11 ground stations, the total access time increases to 7939 minutes, or 132 hours. This is a 34-hour increase over the semi-synchronous orbit which had been a significant increase over the low-earth orbit. So far, the Molniya orbit offers the longest total access times of any orbit examined to this point.

This orbit has demonstrated to this point that it has the access time to the ground stations to be economically viable. Also, due to the orbit type, it would prove to be a much better proof-of-concept orbit, as the access time well exceeds the time not in access. There are some problems with this orbit however. One is that perigee is very close to the Earth, meaning that the vehicle will be subjected to atmospheric stresses. A large vehicle will have trouble maintaining this orbit without significant orbit

adjustments.

Another problem with this orbit is that apogee lies between the semi-synchronous and geosynchronous orbits. This places apogee well beyond most of the satellites in orbit, which significantly increases the chance that another satellite will be radiated or lased, just as with the semi-synchronous orbit. Overall, however, this orbit does provide a balance between access time and orbital problems. It would provide a good proof-of-concept design but would easily scale up with more ground stations and easily become fully operational and potentially economically viable.



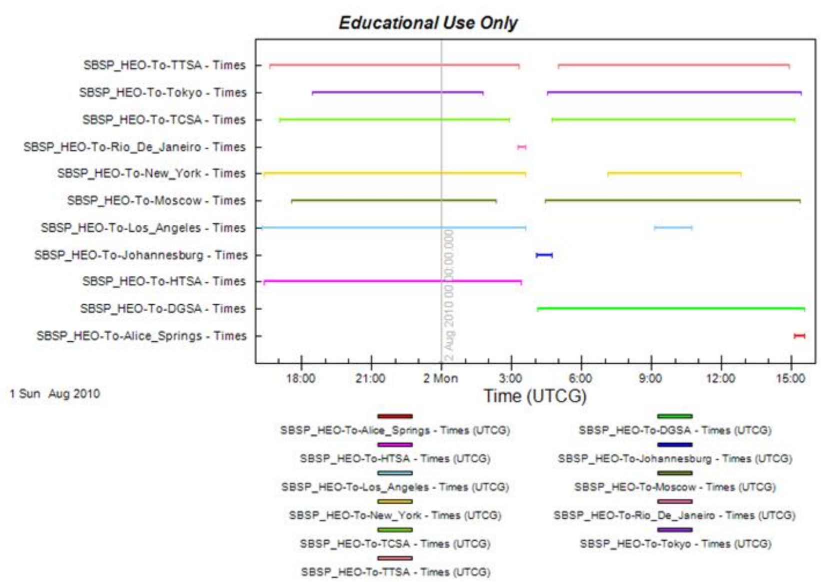


Figure 12. Molniya or HEO Access Time

4. Geosynchronous Orbit (GEO)

The final orbital discussion is a geosynchronous orbit. This type of orbit has the potential to offer constant access to multiple ground stations depending on the location of the ground stations and the type of geosynchronous orbit. Also, since the vehicle has constant access to ground stations, it makes for the perfect technology demonstration and is also easily scaled up to a fully operational program.

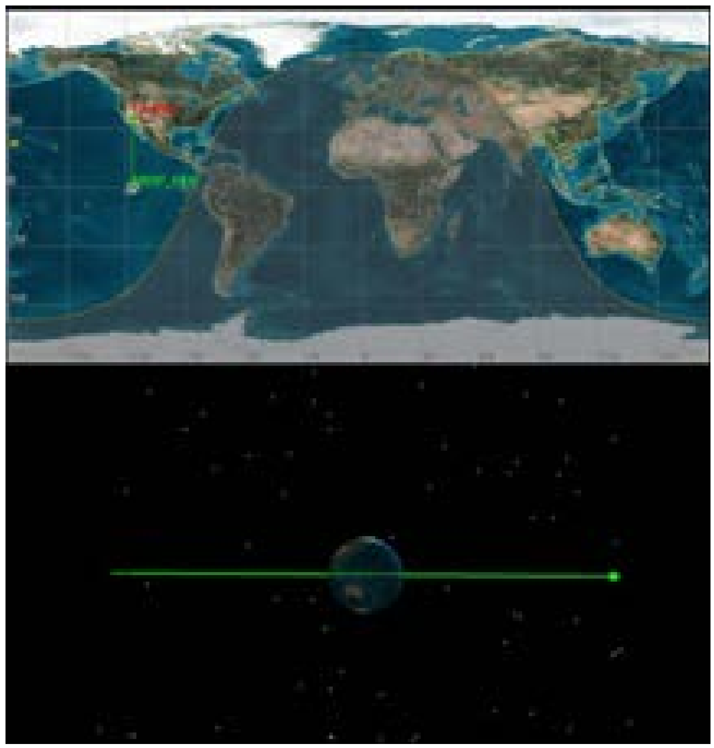


Figure 13. Geosynchronous Orbit

This geosynchronous orbit is, in actuality, a geostationary orbit. It is at zero degrees inclination at an altitude of 35,788 kilometers. This places the vehicle directly over the equator at a point that changes only slightly over time. This placement is what offers the constant access to ground stations. For this orbit, there is no eccentricity. This means that there is no apogee or perigee. Because there is no apogee or perigee and there is no inclination, the vehicle's position must be described by true longitude. True

longitude is the “angle from the principal direction to the spacecraft’s position” (Sellers, 2000, p. 165). For this orbit, the true longitude is 265º E, or 95º W. This places the vehicle over the middle of the inhabited portion of the western hemisphere, allowing for the widest view of the western hemisphere.

The vehicle has been placed within sight of a single ground station in the Mojave Desert in a demonstration mode. As with the other orbital discussions, this was done to see if this orbit would show the cost-effectiveness of a system based on a single vehicle and a single ground station. Then, staying in the same location, the full complement of 11 ground stations were added to observe the vehicle in the fully operational configuration.

After running the accesses on the vehicle in both configurations, the results show that the vehicle has constant access to each ground station in the western hemisphere. For this type of system, constant access to the ground stations while the satellite is illuminated by the sun is the optimal situation. This allows the vehicle to transmit energy, whether by laser or microwave, all the time. A system with a vehicle in a geostationary orbit will minimize the cost-per-kilowatt hour due to the constant energy transmission.

Access	Start Time (UTCG)	Stop Time (UTCG)	Duration (sec)	Duration (min)
1	7/8/2010 16:00	7/9/2010 16:00	86400	1440

Table 14. Geosynchronous Orbit Access to Single Ground Station

The access times shown in Tables 14 and 15 tell much the same story. The vehicle is in constant access with any visible ground station all the time. Even though a vehicle in a geostationary orbit only has access to five ground stations, an interesting fact appears after reading the tables; the vehicle has more access time than every orbit except the Molniya orbit. That difference is about 7,000 minutes or 116 hours, which is significant but very interesting given that the Molniya orbit was calculated using all 11 ground stations but the geostationary orbit only used five.

Total Access Time	Duration (sec)	Duration (min)
	432000	7200
Average	86400	1440

Table 15. Geosynchronous Orbit Access to Five Ground Stations

This information demonstrates that a geostationary orbit is very cost-effective based on the access time and the amount of energy that can be transferred to the ground from the vehicle. Also, because multiple ground stations are in view of the vehicle, it would be easy for the vehicle to transition from one ground station to another if one is unable to receive based on weather or maintenance constraints. Another significant advantage of a geostationary orbit is that, due to the orbit, the vehicle can transmit constantly, meaning that even when the ground station is in the dark, it will still receive energy.

There are disadvantages to a geostationary orbit, however. The first, and potentially most important, is that launching a vehicle to geostationary orbits is the most expensive of all the orbital regimes and also carries a strict weight limit per launch vehicle, depending on which launch vehicle is chosen. It is possible that the weight would be so constrained or the vehicle so heavy that it would take multiple launch vehicles to get the entire satellite into orbit which means that it would have to be assembled on-orbit rather than on the ground. Another disadvantage, as stated in Section II, is that a geosynchronous satellite is above the orbits of approximately 60 percent of operational satellites. Since this vehicle is based on high-energy transmission, there is a significant possibility of the vehicle inflicting unintended harm on another satellite. Finally, a space based solar power satellite operating in a geostationary orbit will require an absolutely perfect pointing accuracy. At the operational distance of 35,788 kilometers, a fraction of a degree in pointing error on the spacecraft will translate to kilometers of error on the ground, resulting in the energy transmitter sending energy to an area not prepared to receive it, which in the case of the laser transmitter, could produce disastrous results.

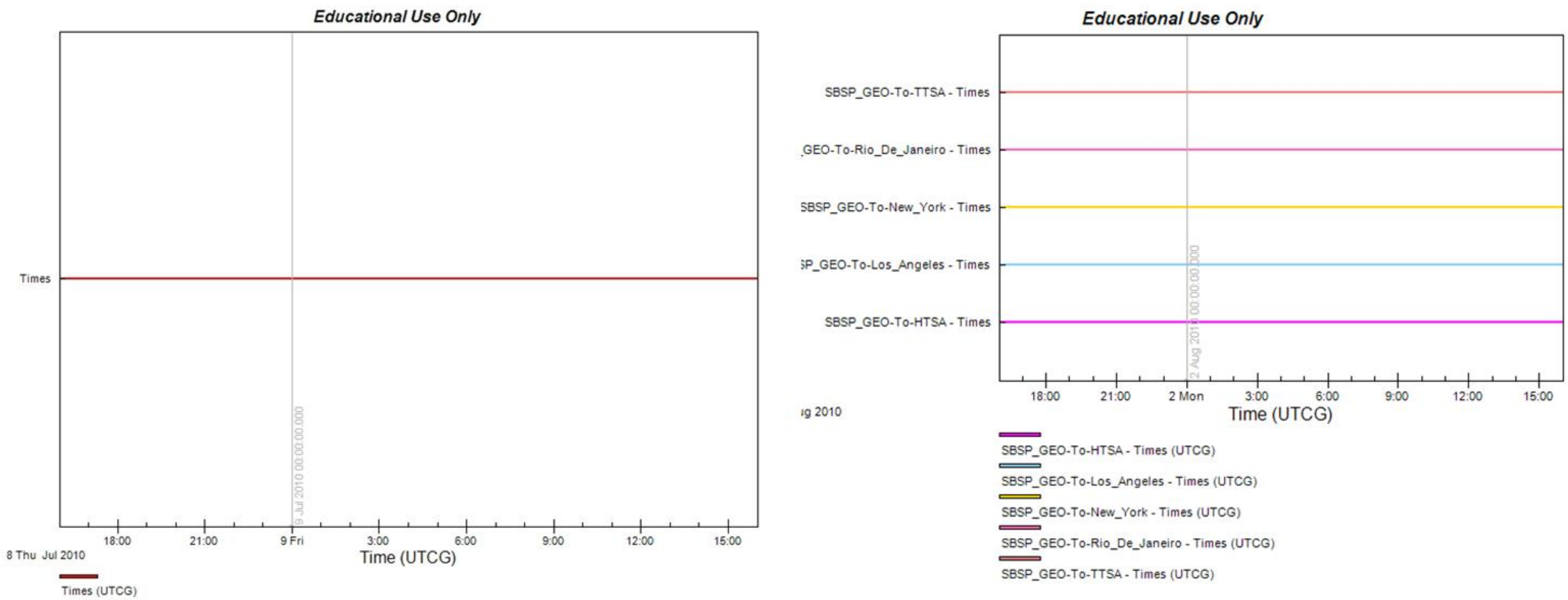


Figure 14. Geostationary Orbit Access Time

C. ORBIT DECISION

After reviewing the four potential orbital regimes, it is time to make a decision on which orbit the vehicle should operate in. Without this decision, it will be difficult to design the remainder of the vehicle. Therefore, weighing the positives and negatives of each orbit is vitally important to coming to a final decision, even though that decision may have negatives that could impact operations.

As described above, a low-earth orbit has the lowest cost-to-weight ratio of any orbit, especially since more vehicles are available to launch into that type of orbit. It is also at or beneath a great deal of satellites in orbit today, meaning that it will not likely radiate or lase other vehicles when transmitting. A distinct downside to this type of orbit is the length of access to a ground station. A low-earth orbit will only dwell above a target for 12–15 minutes, as shown in Table 8. This does not leave much time to transmit energy to a ground station. Another problem of low-earth orbits is the amount of drag experienced by the vehicle. It generates a lot of aerodynamic force on the vehicle, especially one of this potential size.

A semi-synchronous or middle earth orbit increases the dwell time over the target from 12–15 minutes to approximately 300 minutes, according to Table 11. This increase in dwell time is critical for an energy transfer system. It will, however, cost more to put a vehicle into a semi-synchronous orbit than it would into a low-earth orbit. The aerodynamic forces exerted on the vehicle are significantly decreased, which is good for this vehicle. Another problem of this vehicle is that it is above a good portion of the vehicles in orbit, increasing the likelihood of radiating another vehicle.

The Molniya, or highly-elliptical orbit, is designed to have long dwell times over a specific hemisphere and it performs as advertised. It has the longest total dwell time over ground stations of all four orbital regimes. In the example presented, for ground stations in the northern hemisphere, this is a highly desirable orbit because it offers near-constant access to the vehicle (500–600 minutes out of approximately 700 minutes). However, sites in the southern hemisphere will be severely lacking for access time. However, this orbit could also be tuned to provide the same access to the southern

hemisphere, albeit at the cost of northern hemisphere access. At perigee, the vehicle will be moving extremely fast, increasing the aerodynamic forces on the vehicle significantly. This orbit does not have the lowest cost-to-weight ratio into orbit, but it also does not have the highest.

Finally, the geostationary orbit provides constant access from 35,788 kilometers. However, depending on the true longitude of the vehicle, only a few ground stations will be in view of the vehicle. So, while the access is constant, it is only for a few ground stations, not all of them. It would take multiple vehicles in a geostationary orbit to cover all 11 ground stations. An advantage of this orbit is the non-existent aerodynamic force applied to the vehicle. There is essentially no atmosphere at this altitude, so there is very little to no applied aerodynamic force on a vehicle in a geostationary orbit. There are also two very significant disadvantages to this orbit. First, it costs a lot of money to put any vehicle into geostationary orbit. There are also significant limits on a payload, including mass as well as volume. Second, every satellite in orbit that is not in a geosynchronous orbit (558 satellites) will, at some point, come between this vehicle and the ground stations, so there is a significant chance of radiating another satellite. Finally, as discussed previously, the pointing accuracy of a space based solar power satellite operating in a geostationary orbit needs to be absolutely perfect in order to avoid an accidental radiation of an area not prepared to receive an energy transmission.

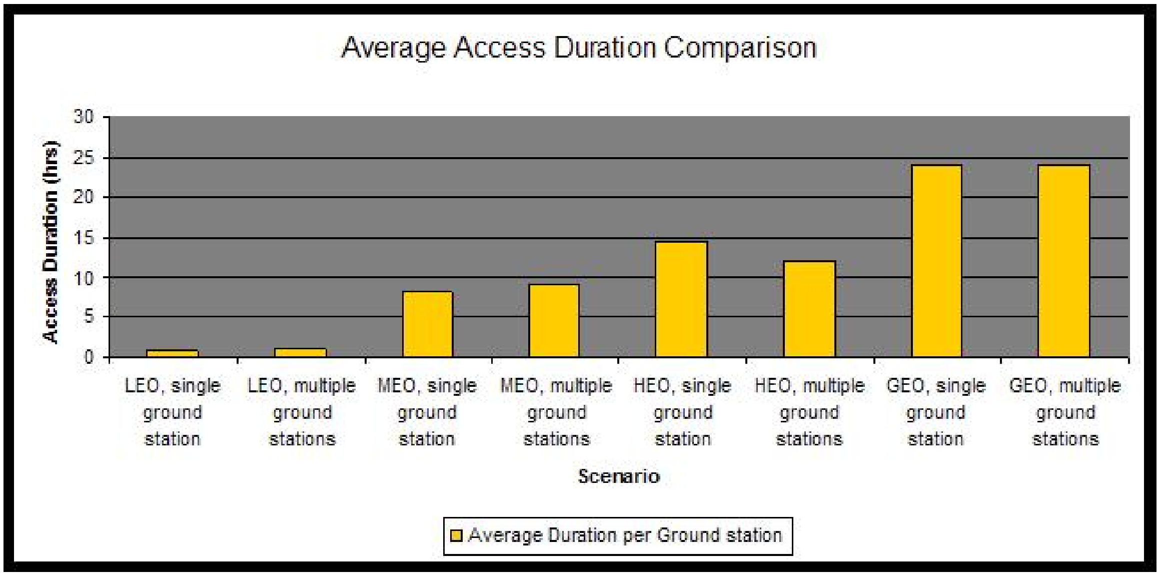


Figure 15. Average Access Duration Comparison

One way to help make a decision is to look at the access times. Figure 15 shows the duration of the average access depending on the scenario. The orbits are in pairs, with one bar representing the single ground station scenario and another bar representing the multiple ground station scenario. Almost all of the figures remain the same or increase from the first to the second scenario with the exception of the Molniya orbit. That is because the ground stations in the southern hemisphere will have very short durations while the ground stations in the northern hemisphere will have very long durations. This difference causes the average to decrease over a single ground station in the northern hemisphere.

Metric	Duration (sec)	Duration (min)
LEO		
Minimum	43.68	0.73
Maximum	921.65	15.36
Average	3719.29	61.98
MEO		
Minimum	320.12	5.34
Maximum	44777.49	746.29
Average	32296.4	538.27
HEO		
Minimum	1095.71	18.26
Maximum	41175.30	686.26
Average	43303.37	721.72
GEO		
Minimum	86400	1440
Maximum	86400	1440
Average	86400	1440

Table 16. Access Duration Metrics by Orbit

Another way to draw comparisons is to look at the minimum, maximum and total access time. Table 16 displays these three metrics for all four orbits. As the table shows, the minimum access duration for the low-earth orbit and the semi-synchronous orbit are very short, but those minimums for both the highly elliptical and geostationary orbits are more than long enough to permit a significant energy transfer. The maximums of all four orbits permit significant energy transfer, but the semi-synchronous, highly elliptical and geostationary orbits are very long. Finally, the total duration of access for all four orbits is more than long enough to support an energy transfer system, but the highly elliptical and geostationary orbits well surpass the other two orbits.

Finally, to make an informed decision, it is important to view the critical factors that have been discussed to this point in chart form. This chart will provide all required information in a summary so that comparisons can be made from orbit to orbit. In this chart, risk factors have been assigned to each orbit and each factor based on the discussion in the previous portions of this section. The chart is displayed below as Table 17.

Orbit Type	Cost/Weight Ratio	Dwell Time	Aerodynamic Forces	Satellites Beneath Orbit
LEO	Low	**Short**	**High**	Few
MEO	Moderate	Moderate	Low	Some
HEO	Moderate	Long (Northern Hemisphere)	High (at Perigee)	Many (at Apogee)
GEO	**High**	Constant	Very Low	**Almost All**

Table 17. Critical Factors and Associated Risks

Based on all of the access times as shown in Table 16 and the information presented in Table 17, a decision on the final orbit can be made. For the constant dwell time over the ground stations and the low risk to the vehicle itself in the orbit, the geostationary orbit will be selected for this system architecture. A geostationary orbit will provide an excellent location for a system demonstrator as well as the fully operational system since it provides constant, 24-hour access to a ground station whether there is one or five ground stations. Operating in GEO places very low stress on the vehicle due to the altitude at which the satellite operates. A significant hurdle with this architecture will be the high cost-to-orbit for the vehicle, the weight limits placed on the launch vehicles going out to a geosynchronous transfer orbit, and the extreme pointing accuracy required for the beamed energy. If the vehicle is too large, assembly will have to be done in orbit which may not be easy. Also, the altitude could cause a problem due to the number of satellites operating between the vehicle and the ground stations as well as the sheer

distance the transmitted energy will have to traverse. In addition, due to the high altitude of a geostationary orbit, a perfect pointing accuracy is required to ensure the energy is transmitted to the correct location on Earth; a small error on the vehicle will induce a large error on the Earth. However, these problems can be overcome with the proper application of technology, whereas the access time is dependent on only the orbit type. These factors make the geostationary orbit the best choice for this system architecture.

D. CHAPTER SUMMARY

This chapter was dedicated to selecting an operational orbit for the space-based solar power system. Four different orbits were discussed: a low-earth orbit, a semi-synchronous or middle earth orbit, a Molniya or highly-elliptical orbit and a geosynchronous orbit. For each orbit, vehicle accesses were calculated against two separate scenarios, a single ground station representing a potential demonstration program and a full complement of 11 ground stations, based upon the assumptions made in Chapter II.

The low-earth orbit was the cheapest to launch into and had the lowest probability of radiating another satellite, but had the shortest dwell time and induced large stressors on the vehicle due to atmospheric drag. The semi-synchronous orbit cost more to launch into orbit, but significantly increased the vehicle’s dwell time over the ground station. However, it increased the probability of radiating another satellite. The Molniya orbit provided long dwell times for ground stations in the northern hemisphere, but sacrificed access time to stations in the southern hemisphere. While the overall access time increased over both the low-earth and semi-synchronous orbits, only the northern hemisphere sites saw an increase. Also, the altitude and speed of the vehicle at perigee would put large stressors on the vehicle. Finally, the geostationary orbit offered constant, 24-hour-a-day access to ground stations within view. This limited the number of ground stations from the assumed 11 down to five. Those five ground stations, however, could receive transmitted energy constantly. The cost to launch a vehicle into geostationary orbit is very high, there are limits to the weight and size that the launch vehicle can carry,

and there is a requirement for perfect pointing accuracy. This could cause the vehicle to have to be assembled on orbit which presents numerous engineering challenges.

In the end, it was mostly the constant access to the ground stations that influenced the decision to select a geostationary orbit for this architecture. Constant access, coupled with very low stressors on the vehicle, is a favorable situation especially if the vehicle is large. As stated in the orbit selection portion of this chapter, the cost, assembly and potential vehicle radiation can be solved with science and engineering while Newton and Kepler cannot. The selection of the geostationary orbit takes into account the factors that can be changed given the proper amount of forethought and hard work and balances those with the factors that can never be changed because they are a simple fact of nature.

Orbit selection was an important piece of the puzzle for this system architecture. Without the operational orbit in place, the vehicle, and more specifically the payload, cannot be designed. Now, with the orbit in place, decisions can be made on the vehicle. The following section will examine the vehicle specifications, given the orbit and the amount of energy the vehicle will have to transmit in order to be cost-effective. This will involve looking at existing satellite busses, designing the energy transmitting payload and selecting the launch vehicle (or launch vehicles) to arrive at a final system cost. Then, based on the amount of energy the payload can transmit, a cost-per-kilowatt can be calculated to compare with the current fossil fuel systems in use. None of the following discussions would be valid, however, if it were not for the orbit selection undertaken in this chapter.

IV. SPACECRAFT DESIGN

A. INTRODUCTION

Now that the orbit has been selected, the spacecraft design phase can begin. It was critical to establish the orbit first in order to understand the weight problems associated with launch operations as well as the energy transmission distances as it pertains to the payload design. This section includes discussions about the general system power requirements, the design of the solar array, the spacecraft bus and, most critically, the power transfer subsystem. The power requirements will be derived from the previously established worldwide power requirements. From that requirement, the solar array can be designed based on current technology. There will also be related discussions about possible changes based on technology advances and how that could influence the solar array design.

To date, there are numerous satellites that operate in geosynchronous orbit. The demand for these types of satellites, often communications or weather satellites, has generated a significant commercial market for off-the-shelf spacecraft bus designs that operate in this orbital regime. Often, they are three-axis stabilized, can handle high power loads, have large solar arrays to power the spacecraft, and contain the amounts of propellant necessary to maintain a geosynchronous orbit over the design life of the spacecraft. Due to this large commercial market, it would make the most sense to integrate a commercial, previously designed bus into the satellite. Three of the most common commercial geosynchronous satellite buses are examined in detail later in this section.

Finally, this section examines the two competing designs in the research and development community for the energy transfer payload: a microwave-based system and a laser-based system. As discussed earlier in this thesis, both types of payloads have their advantages and both have their challenges. Ultimately, it is the trade-off between the capability of the payload and the amount of time and work required to defeat the challenges that will determine which payload is selected for the final spacecraft design.

B. SYSTEM POWER REQUIREMENTS

To begin the design of the satellite bus and payload, the system's baseline power requirement must be set. The power generation requirement is inherently tied to the size of the solar panel which impacts the spacecraft weight and, therefore, cost for the launch vehicle. In order to have a meaningful discussion on the viability of a space-based solar power architecture, it is critical to note that the system does not need to replace all of the fossil fuel energy generation nor does it have to achieve a certain percentage of all generated power. The critical piece to the overall equation of viability is the cost per kilowatt. It is not currently important to know the total cost of the mission, but rather it is important to understand the relationship between the generated power and the cost drivers. These drivers include the cost of design, construction and integration of the satellite itself, the launch vehicles necessary to place the satellite into orbit, the operations and maintenance budget necessary to keep the satellite operational and the operations and maintenance budget necessary to keep the ground systems operational. With those kinds of costs involved, the satellite inherently needs to generate a significant amount of power in order to remain competitive in the cost per kilowatt category.

Based on research conducted by the National Security Space Office in 2007, a space based solar power satellite should have an operational goal of "delivering 1–10 GWe (p. 31)" to Earth. This is the performance characteristic that will drive the remainder of the satellite design discussion. For the sake of simplicity, the power delivery metric will be 1GWe. This will establish the low-end requirements as seen by the NSSO in 2007. From there, the system can be scaled based on the required GWe metric.

Ultimately, regardless of the payload selected for integration into this system, there will be inefficiencies generated by various internal and external factors that impact the amount of energy the satellite delivers to the ground. For example, a laser system has an inherent inefficiency in turning electricity into light for transmission to the ground. In some cases, like in the case of an electrical diode pumped laser currently under development at Lawrence Livermore National Laboratory, that conversion efficiency is 50 percent; for every 1 GWe transmitted to earth, the input power into the laser would have to be 2 GWe (Rubenchik, Parker, Beach, & Yamamoto, 2009, p. 3).

Microwave emitters, on the other hand, do not suffer from the same conversion inefficiencies. These systems have “high conversion efficiencies in space and on the ground, with good transmission through the atmosphere, even during periods of heavy cloud cover” (Rubenchik et al., 2009, p. 9). Overall, based on the NSSO 2007 report, a microwave-based payload has a transmission efficiency between 80 percent and 90 percent (p. 21). In order to build a payload-agnostic system, the system power requirement prior to any interaction with the transmission payload will be based on the least efficient transmission system. In this case, the laser payload with a 50 percent efficiency rating establishes the ultimate threshold for the system. Based on the previously discussed 1 GWe delivery requirement, the solar panels must provide 2 GWe to the payload.

C. SOLAR ARRAY DESIGN

With the system power requirement now established, the power generating solar panels can be designed based on the current and anticipated materials and the known mathematical equation for solar array size based on material efficiency. Prior to the rest of the solar panel discussion, it is important to note that this thesis will concentrate on a traditional solar panel, relying on linear panels of photovoltaic cells to generate electricity from sunlight.

Recently, however, the generic space based solar power satellite design has featured mirrors that concentrate sunlight onto the solar panel, thereby increasing the efficiency of the solar panel while decreasing the weight and size of the panel itself. The case provided by Rubenchik et al. (2009) states that “under 300-sun concentration, 1 cm of solar cell area produces the same electricity as would 300 cm^2 without concentration (p. 8).” Being able to generate the same level of electricity with 300 times less solar panel area could have a huge impact on the size and launch capability of a space based solar power satellite. While the design of the solar panels is not based around this configuration for the purposes of this thesis, there will be a brief discussion on the potential impacts at the close of the section on array sizing for comparison.

1. Materials Discussion

As stated in Space Mission Analysis and Design (SMAD), there are currently five different types of photovoltaic solar cells that are addressed in the book specifically. They are presented below in Table 18, as seen in SMAD. Associated with all of the different types are the theoretical peak efficiency, the current best achieved efficiency (both in production and in the laboratory) and the equivalent time in geosynchronous orbit for 15 percent degradation. In addition to the traditional solar cells provided by SMAD, the National Renewable Energy Laboratory has developed a new solar cell called a concentrator photovoltaic (CPV) cell. According to Rubenchik et al. (2009), "this thin, lightweight cell will transform concentrated solar radiation into electricity with an efficiency of approximately 40 percent (p. 7)." While this satellite is not being designed as a concentrator system, the existence of this "high efficiency" cell establishes a current technological threshold from which to draw conclusions on the viability of the system.

Cell Type	Silicon	Thin Sheet Amorphous Si	Gallium Arsenide	Indium Phosphide	Multi-junction GaInP/GaAs	Concentrator Photovoltaic
Planar Cell Theoretical Efficiency	20.8%	12.0%	23.5%	22.8%	25.8%	40%
Achieved Efficiency						N/A
Production	14.8%	5.0%	18.5%	18%	22%	
Best Laboratory	20.8%	10%	21.8%	19.9%	25.7%	
Equivalent time in geosynchronous orbit for 15% degradation						Data Not Available
1MeV electrons	10 yr	10 yr	33 yr	155 yr	33 yr	
10MeV protons	4 yr	4 yr	6 yr	89 yr	6 yr	

Table 18. Performance Comparison for Photovoltaic Solar Cells (From Wertz & Larson, 1999, p. 414)

As with any technological decision, there are multiple factors that enter into the ultimate design decision. These factors can include size, weight, cost, and technology readiness. All of these different cell types listed in Table 18 have different factors associated with them. For example, according to Wertz and Larson (1999), “silicon presently costs the least for most photovoltaic power applications, but it often requires larger area arrays and more mass than the more costly gallium-arsenide cells. Programs for which mass and volume (solar array area) are critical issues may allow higher costs or technical risks. They could select a system based on gallium arsenide or some other advanced type of solar cell. Risk develops from the unproven reliability and fabrication of the photovoltaic source” (p. 413).

For this system, the solar array is going to be significantly larger than a standard solar array. The actual size required for each type of cell is calculated and compared in the next subsection. However, based on the six types of cells presented above in Table 18 and the data provided in the above paragraph, a few immediate decisions become clear. The first is that, based on Wertz and Larson's assessment, silicon-based cells are not likely to support the size and mass requirements of this system, even if they tend to be cheaper. By the same token, the relatively unproven Concentrated Photovoltaic Cell (CPV), while it could save significant weight and array area, would add significant risk and add to the already-large budget.

2. Array sizing

To make the final design decision on the solar panels, the last calculation is the solar array size. This is based on the efficiencies of the solar panels as described in the previous subsection. The equation to calculate the required solar array area is found in *Spacecraft Mission Analysis and Design (SMAD)*, which states that:

$$A_{sa} = \frac{P_{sa}}{P_{EOL}}$$

where A_{sa} is the area of the solar array, P_{sa} is the system power requirement, and P_{eol} is the "array's performance per unit area at end-of-life" (Wertz & Larson, 1999, p. 417). To begin the area calculation, the first equation is for the system power requirement, P_{sa}:

$$P_{sa} = \frac{\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}}{T_d}$$

This equation determines the system power requirement based on the power necessary to operate during both eclipse and daylight (P_e and P_d respectively), and the lengths of both eclipse and daylight per orbit (T_e and T_d). It also takes into account "the efficiency of the paths from the solar arrays through the batteries to the individual loads and the path directly from the arrays to the loads" (Wertz and Larson, 1999, p. 413). These terms are called X_e and X_d, respectively. "The efficiency values for eclipse and daylight depend on the type of power regulation: direct energy transfer or peak-power

tracking. For direct energy transfer, the efficiencies are about $X_e = 0.65$ and $X_d = 0.85$; for peak-power tracking they are $X_e = 0.60$ and $X_d = 0.80$. The efficiencies of the former are about 5 percent to 7 percent greater than the latter because peak-power tracking requires a power converter between the arrays and the loads" (Wertz & Larson, 1999, p. 413). With the critical function that this solar array accomplishes, it should be designed with peak-power tracking, meaning that it will be slightly less efficient but will have the inherent ability to track the peak-power.

According to the 2007 NSSO report, "a special case occurs at Geostationary (GEO) orbit where the orbital period of the satellite corresponds to the speed of Earth's rotation and the satellite appears stationary over the ground. In this location, very little beam steering is required, and due to the axial tilt of the Earth with respect to the Sun, spends less than 1 percent of the total time in shadow" (p. A-2). With that information in mind, the original equation for the solar panel array becomes simpler. The original term:

$$\frac{P_e T_e}{X_e}$$

goes to zero because the satellite effectively spends no time in eclipse. This makes the simplified equation:

$$P_{sa} = \frac{\frac{P_d T_d}{X_d}}{T_d}$$

The next piece of the solar array area calculation is the equation for end-of-life system power (P_{EOL}). The equation for P_{EOL} is illustrated again by Wertz and Larson (1999, p. 417):

$$P_{EOL} = P_{BOL} \times L_d$$

This equation introduces two new terms: the system beginning-of-life power (P_{BOL}) and the lifetime degradation (L_d). The system beginning-of-life power is captured in SMAD on page 417:

$$P_{BOL} = P_0 \times I_d \times cos\theta$$

where, according to Wertz and Larson, "P_0 is the estimated power output of the solar cell with the Sun normal to the surface of the cells" (p. 412), I_d is the inherent degradation due to "design and assembly, temperature of array, and shadowing of cells" (Wertz & Larson, 1999, p. 414). Finally, "cosθ is referred to as the cosine loss. We measure the Sun incidence angle, θ, between the vector normal to the surface of the array and the Sun line. So if the Sun's rays are perpendicular to the solar array's surface, we get maximum power" (Wertz & Larson, 417).

The final piece of this puzzle is the lifetime degradation, which is calculated using the following equation:

$$L_d = \left(1 - \frac{degradation}{year}\right)^{satellite\ life}$$

"Life degradation, L_d, occurs because of thermal cycling in and out of eclipses, micrometeoroid strikes, plume impingement from thrusters, and material outgassing for the duration of the mission. In general, for a silicon solar array in LEO, power production can decrease by as much as 3.75 percent per year, of which up to 2.5 percent per year is due to radiation. For gallium-arsenide cells in LEO, the degradation is about 2.75 percent per year, of which radiation causes 1.5 percent per year" (Wertz & Larson, 1999, p. 417). Finally, multijunction solar cells experience degradation on the order of 0.5 percent per year (Wertz & Larson, 1999, p. 412).

To begin calculating the required solar array area, the first necessary step is to calculate the amount of power the solar array is going to be required to produce. As stated before, that equation is now:

$$P_{sa} = \frac{\frac{P_d T_d}{X_d}}{T_d}$$

The power required in daylight was established earlier at 2 GWe. The time in daylight (T_d), also equivalent to the full orbital period, is 86,164.0905 seconds. As discussed earlier, X_d=0.80. Using the appropriate substitutions:

$$P_{sa} = \frac{\frac{(2 \times 10^9 W) \times 86164.0905s}{0.80}}{86164.0905s} = 2.5GW$$

The next required calculation is to determine the beginning of life power for the solar cells. According to Wertz and Larson (1999), P_0 varies based on the type of solar cell. “A silicon cell generates a maximum power of 202 W/m2. A gallium-arsenide cell generates 253 W/m2. Finally, a multi-junction cell generates 301 W/m2” (p. 412). Taking that information, the substitutions into the equation for P_{BOL} appear as:

$$P_{BOL} = P_0 \times 0.77 \times cos0$$

The resulting figures for each type of cell are:

Silicon	155.54 W/m^2
Gallium Arsenide	194.81 W/m^2
Multi-junction	231.77 W/m^2

Table 19. Solar Array Beginning-of-Life Power

Now, the life degradation needs to be calculated in order to determine the end-of-life power. Using the original equation and using a 10 year design life:

$$L_d = \left(1 - \frac{degradation}{year}\right)^{satellite\ life}$$

which results in life degradation figures as shown below:

Silicon	0.6823
Gallium Arsenide	0.7566
Multi-junction	0.9511

Table 20. Solar Array Life Degradation

Now that the beginning-of-life power and the life degradation have been calculated, the end-of-life power can be calculated. For the three types of solar cells, this results in a P_{EOL} of:

Silicon	106.12 W/m^2
Gallium Arsenide	147.39 W/m^2
Multi-junction	220.43 W/m^2

Table 21. Solar Array End-of-Life Power

Finally, with all of the required information calculated, the solar array area can be calculated. Using the original equation and substituting the numbers based on the calculations just completed, the solar array areas required to support a 2GWe power transmission payload are captured in Table 22.

Cell Type	P_{BOL}	Life Degradation	P_{EOL}	A_{sa}
Silicon	155.54 W/m2	0.6823	106.12 W/m2	2.3558x10^7 m2
Gallium Arsenide	194.81 W/m2	0.7566	147.39 W/m2	1.6961x10^7 m2
Multijunction	231.77 W/m2	0.9511	220.43 W/m2	1.1341x10^7 m2

Table 22. Solar Array Size Calculations

To roughly estimate the same calculation for the concentrated photovoltaic cell, there is a conversion factor presented in the paper by Rubenchik et al. (2009). In that paper it is stated that “under 300-sun concentration, 1cm^2 of solar cell area produces the same electricity as would 300cm^2 without concentration (p. 8). Rubenchik et al. (2009) also describes the CPV cells as multi-junction cells (p. 8). Therefore, the multi-junction solar array area presented above in Table 22 will stand-in for the CPV cell, with the understanding that the CPV cell has an efficiency approximately 10 percent higher than the best laboratory multi-junction cell, which would further reduce the required solar array area. To calculate the estimated solar array size for the CPV cell, a simple ratio can be used. After re-arranging the ratio to calculate the A_{CPV}:

$$A_{CPV} = \frac{\left(0.01m^2 \times A_{MJ}\right)}{3m^2} = \frac{\left(0.01m^2 \times (1.1341 \times 10^7)\right)}{3m^2} = 37803.33m^2$$

This is a significant size decrease over the unconcentrated cell type. That type of size decrease also has a significant effect on the mass of the system, potentially allowing for decreased launch cost. There are other pieces to the concentrator system puzzle, the least of which is the size and weight of the mirror required to focus the Sun on the concentrator solar panel. This type of system will be described in greater detail in the remaining sections.

The final piece of the solar array puzzle is the weight of the array. The weight of the notional array will provide a rough estimate of the launch vehicle or vehicles necessary to launch this system into orbit which will, in turn, provide a very rough estimate on the cost-to-orbit. Wertz and Larson (1999) provides some insight into a general calculation using the equation found on page 333:

$$M_a = 0.04P$$

where Ma is the mass of the array and P is the power output, in watts, of the array. Using this equation, the mass of the array is roughly 100,000,000 kg.

D. SPACECRAFT BUS AND PAYLOAD

The next critical piece of spacecraft infrastructure to design is the spacecraft bus and the associated subsystems. The following subsections examine a few choices for commercial satellite bus that could support the power transfer payload and the solar array. It would be more cost effective to utilize an existing bus rather than design and build a bus from the ground up. The next subsections also look in-depth at the two main options for the power transfer payload and identify all of the system requirements associated with the power transfer payload.

While the solar array sizing was a critical piece of the system puzzle, the spacecraft bus has the potential to cut the program budget significantly by utilizing a bus that benefits from economies of scale. There are also two potential technologies vying for the power transfer payload: a microwave-based system and a laser transfer payload. Both of these payloads will be examined in-depth for their suitability for inclusion in this system.

1. Spacecraft Bus

A spacecraft of this size and complexity is going to require a significant spacecraft bus to support the advanced payload and solar array. With the high cost associated with the payload and solar array, it could be a cost-saving measure to use a commercial spacecraft bus designed to operate in a geosynchronous orbit. These existing satellite systems would benefit from economies of scale and pre-existing production

lines. There are three more commonly used spacecraft buses in production: the STAR geostationary satellite built by Orbital Sciences, the Boeing Company's 702 geostationary satellite, the A2100 geostationary satellite built by Lockheed Martin Company, and the Mitsubishi DS2000 geostationary satellite.

Each of these buses have varying infrastructures depending on their intended use. The following subsections detail each bus individually for their suitability as the backbone of this space based solar power system. Following these three subsections, Table 23 will compare all four buses side-by-side on the critical design parameters. It is important to note during this examination that the satellite bus selected here would provide a housing for the power transmission payload and support the rest of the system with the necessary communications and station-keeping functions with one exception. Due to the large solar array, it is assumed that the bus will not provide direct physical support to the array. Instead, the array will be tethered but remain physically separate from the bus.

When determining which bus is more suitable for use in this system, it is important to consider what component requirements exist in order to support the power transfer mission. These component requirements include a highly stable but mobile bus, a high degree of pointing accuracy, a high mass capacity and a high electrical load capability. Based on the information presented and calculated in the above subsections, it is not expected that any current commercial bus would have the mass or electrical capacity in its stock configuration. That being said, it is critical to examine each candidate bus to determine which is the most suitable for this mission.

a. Orbital Sciences STAR Geostationary Satellite

"Orbital's STAR GEO platform is the right-sized answer for your satellite mission needs with proven reliability and a wide range of payload capabilities. The industry's top operators have been relying on the STAR platform for their small to medium missions (up to 5kW payload) for over a decade. And with the introduction of our more powerful, next generation STAR-2 high power bus, Orbital STAR satellites now offer even greater mission flexibility" (Orbital Sciences Corporation, 2009, p. 2).

“The STAR Bus satellite platform is a modular, mass efficient structure, designed for simplified integration to reduce manufacturing cycle times. The satellite bus consists of three major elements: mechanical, power, and command/telemetry subsystems. The strength of the structure is derived from the mechanical design incorporating a composite thrust cylinder, to which the bus, payload, nadir, and base panels are connected through primary and secondary support structures. Energy from two multi-panel solar wings and lithium ion batteries is electronically processed to provide 35V regulated power to the satellite throughout the mission. All active units aboard the spacecraft are connected through a MIL-STD-1553B flight processor. Commands from the ground are processed by the processor or bus controller to the intended unit, and telemetry is collected from the units and sent to the ground station for state-of-health monitoring. The modularity of the structure and the standard digital interfaces of the 1553B allow parallel assembly and test of the bus and payload systems, reducing manufacturing schedule risk by minimizing the time spent in serial satellite integration and test flow” (Orbital Sciences Corporation, 2009, p. 1).

Figure 16. STAR Satellite Bus (From Orbital Sciences Corporation, 2009)

“While primary applications are Fixed Satellite Services (FSS) and Broadcast Satellite Services (BSS), STAR Bus can be adapted for Earth and Space Science applications, as well as for technology demonstration or risk reduction programs. Depending on mission duration requirements, STAR Bus can accommodate payloads in

excess of 500 kg, and provide up to 5000W of power. Instrument data can be provided in standard format such as CCSDS or through secured encryption, as approved by the National Security Agency (NSA)" (Orbital Sciences Corporation, 2009, p. 1).

The STAR bus has flight heritage as the bus for satellites such as the INTELSAT New Dawn and Intelsat missions, the SES Americom AMC satellites, the Telenor THOR 5 satellite, PanAmSat's Galaxy satellite line, and the Optus D-Series satellites. These satellites stretch back to launches back in 1997, so there are significant gains in established production lines and flight heritage.

The STAR Bus itself has an advertised payload mass capability of more than 200 kg, but the Fact Sheet does not establish a maximum weight (Orbital Sciences Corporation, 2009, p. 4). It also is advertised as providing 555W end-of-life power to the payload (Orbital Sciences Corporation, 2009, p. 4). It is a three-axis stabilized, zero momentum bus, so it is very stable and will enable a high pointing accuracy (Orbital Sciences Corporation, 2009, p. 4). It uses a hydrazine monopropellant attitude control system for stationkeeping (Orbital Sciences Corporation, 2009, p. 4). An important piece, however, is that the STAR Bus has "several available options [that] augment the basic bus to provide improved pointing, more payload power, secure communications, higher downlink data rates or enhanced payload computing power" (Orbital Sciences Corporation, 2009, p. 2).

b. Boeing 702 High Power Geostationary Satellite

"The Boeing 702 design is directly responsive to what customers said they wanted in a communications satellite, beginning with lower cost and including the high reliability for which the company is renowned. For maximum customer value and producibility at minimum total cost, the Boeing 702 offers a broad spectrum of modularity. A primary example is payload/bus integration. After the payload is tailored to customer specifications, the payload module mounts to the common bus module at only four locations and with only six electrical connectors. This design simplicity confers major advantages. First, nonrecurring program costs are reduced, because the bus does not need to be changed for every payload, and payloads can be freely tailored without

affecting the bus. Second, the design permits significantly faster parallel bus and payload processing. This leads to the third advantage: a short production schedule" (Boeing Corporation, 2010a, p. 1).

Boeing has also included on the 702HP bus an "advanced xenon ion propulsion system (XIPS), which was pioneered by Boeing. XIPS is 10 times more efficient than conventional liquid fuel systems. Four 25-cm thrusters provide economical stationkeeping, needing only 5 kg of fuel per year – a fraction of what bipropellant or arcjet systems consume" (Boeing Corporation, 2010a, p. 1).

Additionally, according to the Fact Sheet, "the Boeing 702HP also incorporates a bipropellant propulsion system, which can lift the satellite into final orbit after separation from the launch vehicle" (Boeing Corporation, 2010a, p. 1), giving the 702HP bus a significant propulsion system advantage over the other possible systems.

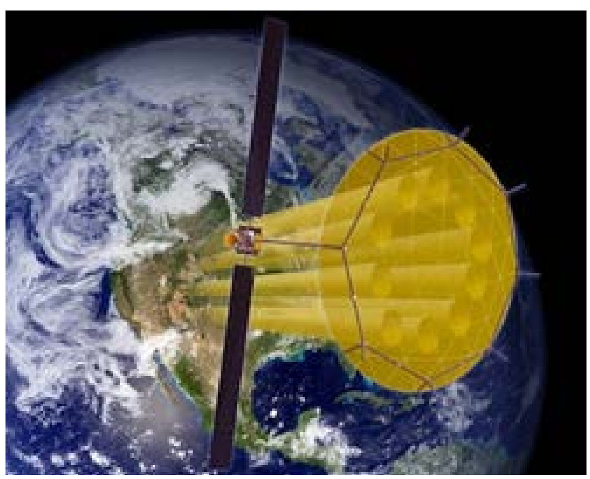


Figure 17. SkyTerra-Boeing 702HP Bus (From Boeing Corporation, 2010b)

Most important to this thesis, "the Boeing 702 offers a range of power up to 18 kW. Dual and triple-junction gallium arsenide solar cells enable such high power levels. Spectrolab, Inc, a Boeing subsidiary, developed the cells" (Boeing Corporation, 2010a, p. 2). It is also critical to understand the possible launch vehicles the bus can use.

"The baseline Boeing 702 is compatible with several launch vehicles. These include the Delta IV, Atlas V, Ariane 5, Proton, and Sea Launch" (Boeing Corporation, 2010a, p. 2).

There is significant production history behind the Boeing 702HP geosynchronous bus. "Boeing announced the innovative satellite series in October 1995, and in 2009 introduced a mid-ranged version, the 702MP for 'mid-power'. At that time the legacy Boeing 702, which has continuously evolved, was designated the Boeing 702HP for 'high-power'. Evolved from the popular, proven 601 and 601HP spacecraft, the body-stabilized Boeing 702 is the world leader in capacity, performance and cost-efficiency. The first Boeing 702HP satellite was launched in 1999" (Boeing Corporation, 2010, p. 1).

Of special note, "in 2006, Boeing received a second major contract from SkyTerra LP to provide two Boeing 702HP satellites, with an option for a third. The satellites will be used to create the world's first commercial wireless communications service, using both space and terrestrial elements" (Boeing Corporation, 2010b, p. 2). These satellites have some of the largest antennas on-orbit, making them specifically relevant to this thesis. "The two satellites are built at Boeing satellite integration and test complex in El Segundo, CA. Harris Corporation of Melbourne, FL, developed the satellites' 22-meter L-band reflector" (Boeing Corporation, 2010b, p. 2).

c. Lockheed Martin A2100 Geostationary Satellite

"From fixed satellite, direct broadcast and IP services, to mobile telephony and broadband, the challenges for commercial and government satellite systems into the 21st century are met by the Lockheed Martin A2100 satellite platform. Modularity at the subsystem and component levels and a broad inventory of fully qualified standard components enable a configure-to-order approach that reduces or eliminates costly non-recurring engineering" (Lockheed Martin Corporation, 2008, p. 1). "Lockheed Martin's A2100 satellites are compatible with Atlas V, Delta, Sea Launch, Land Launch, Ariane 5, Soyuz, H-IIA and Proton launch vehicles. This versatility supports a wide range of missions, while increasing time-to-orbit reliability. Today there are 36 Lockheed Martin commercial A2100 satellites in-orbit, serving customers in all regions of the world. Our

proven performance and the experience we have amassed from numerous customer programs have established the Lockheed Martin A2100 as a satellite platform that can be trusted to complete critical missions" (Lockheed Martin Corporation, 2008, p. 2).

"The A2100 design is highly modular at the subsystem and component levels, and our broad inventory of fully qualified standard components allows a 'configure to order' approach that eliminates costly reengineering. The A2100 design also features a major reduction in parts – simplifying construction, increasing on-orbit reliability and reducing weight and cost. Lightweight all-composite material increase strength, minimize thermal distortions and reduce launch costs. Manufacturing A2100 spacecraft occurs in the Lockheed Martin Commercial Satellite Center. With co-located assembly and test facilities, the center was specifically designed to dramatically cut A2100 production cycles" (Lockheed Martin Corporation, 2008, p. 1).

Figure 18. A2100 Satellite Bus (From Lockheed Martin Corporation, 2008)

d. Mitsubishi Electric DS2000 Geostationary Satellite

"Heeding the call for large-capacity, high-speed communications capable of satisfying the many increasingly diversified needs in the commercial communications satellite market, Mitsubishi Electric developed the DS2000 standard satellite platform.

Decades of participation in satellite projects and the successful results thereof have fostered a wealth of satellite technology within the company, and concerted R&D efforts in the field have brought Mitsubishi Electric to the forefront of the industry" (Mitsubishi Electric Corporation, 2010, p. 1).

Figure 19. DS2100 Satellite Bus (From Mitsubishi Electric Corporation, 2010)

As advertised in the DS2000 fact sheet, the DS2000 can provide 15,000W of power; a significant increase over its contemporaries with the exception of the Boeing 702HP. It has a similar launch vehicle capability: "H2-A, ARIANE-V, DELTA-IV, ATLAS-5, Sea Launch" and others (Mitsubishi Electric Corporation, 2010, p. 1). It is also a three-axis stabilized bus which can be designed as a controlled bias momentum or zero momentum system (Mitsubishi Electric Corporation, 2010, p. 2).

e. Satellite Bus Comparison

Bus Name	STAR	702HP	A2100	DS2000
Payload Power	5 kW	18 kW	12 kW	15 kW
Stabilization	3-axis	3-axis	3-axis	3-axis
ACS	Zero momentum	Zero momentum	Zero momentum	Controlled bias momentum / zero momentum
Launch Capability	Ariane 5, Soyuz, Proton, Land Launch	Ariane 5, Delta-IV, Atlas 5, Proton, Sea Launch	Ariane 5, Delta-IV, Atlas 5, Proton, Sea Launch, Long March	H2-A, Ariane 5, Delta-IV, Atlas 5, Sea Launch
Flight Heritage	High	High	High	Moderate

Table 23. Satellite Bus Comparison

2. Power Transfer Payload

When designing the power transfer payload, there are two leading candidate technologies to choose from: a microwave system and a laser system. Both systems have the capability to transfer 1GWe to a ground site; it is the components within each system that have satellite-wide and system-wide impacts. Each of these systems will be examined in the following sub-sections. Also examined below is the requirement to store power. While a solar power satellite in geosynchronous orbit should be illuminated about 99 percent of the time and should have access to at least one ground site at all times, there may be times where the payload is not able to transmit energy to the surface. With that possibility, it makes good engineering and operational sense to build in storage capacity so collected energy does not go to waste. This capability would most likely exercised during payload transitions from one ground site to another.

a. Power Storage Requirements

For a space based solar power system, the power involved is astronomically higher than a standard spacecraft bus, so the bus as commercially designed should not have to shoulder the load of this payload. Along the same line, it should not have to provide power storage for the payload. For that reason, the payload itself will have power storage built into the design. This allows the batteries to be sized specifically for the payload and will not require a re-design of the bus. According to Wertz and Larson (1999), "energy storage is an integral part of the spacecraft's electrical-power subsystem providing all the power for short missions (< 1 week) or back-up power for longer missions (> 1 week). Any spacecraft that uses photovoltaics or solar thermal dynamics as a power source requires a system to store energy for peak-power demands and eclipse periods. Energy storage typically occurs in a battery, although systems such as flywheels and fuel cells have been considered for various spacecraft" (p. 418).

In the case of this particular system, the power storage is required for short durations over a long mission duration. It is really intended to bridge the gap between power receivers on the ground. As one site becomes unavailable due to weather or maintenance, the payload will have to switch to another site to transfer the power. Since radiating Earth with a microwave emitter or moving a high-power laser over Earth's surface is not a logical course of action, it makes sense to build in storage so that those minutes between receiver sites is not lost.

Ultimately, the design of the power storage system comes down to the number of batteries in the system and the capacity of each battery. To calculate the capacity, there is an equation that Wertz and Larson (1999) presents on page 422:

$$C_r = \frac{P_e T_e}{(DOD)Nn}$$

where P_e is the eclipse power, T_e is the time in eclipse, DoD is the depth of discharge of the batteries, N is the number of batteries in the system and n is the "transmission efficiency between the battery and the load" (p. 422). This system is the temporary backup for the payload, not the bus, so it does not necessarily fit all of the presented

variables. However, due to similarities between the payload and a normal bus, some substitutions can be made. With those substitutions, the equation is:

$$C_r = \frac{P_t T_t}{(DOD)Nn}$$

where P_t is now the power storage necessary during transit (repointing from one ground site to another) and T_t is the time during transit. The other variables remain the same from the original equation.

To calculate the battery capacity, some initial assumptions need to be made. The system is designed to transmit 1 GWe, so a storage system will need to be designed to store that much energy over the period of time it takes the system to acquire a new ground site. Due to the distance from Earth, a solar power satellite will not have to move all that far in order to reposition the transmitter to acquire a new ground site. For this thesis, the period required to move the transmitter to a new ground site is five minutes. This is mostly due to the inherent size of the satellite; moving that much mass will not be easy to start nor stop, so the satellite will have to move slowly to remain accurate and minimize the impact to the system as a whole.

As presented by Wertz and Larson (1999), the depth of discharge (DoD) varies with the type of battery. “For LEO, we expect the battery’s DoD to be 40–60 percent for NiH2 technology, compared to 10–20 percent for NiCd technology” (p. 422). Wertz and Larson (1999) also states that “the geosynchronous orbit demands few charge/discharge cycles during eclipse periods, thus allowing a fairly high (50 percent) depth-of-discharge” (p. 420). Finally, according to Wertz and Larson (1999), “the number of batteries, N, may be equal to one for this calculation if you simply require a battery capacity. Two to five cells are typical. We must have at least two (unless the battery uses redundant cells) because the spacecraft needs redundant operation with one unit failed. But more than five batteries require complex components for recharging” (p. 422). For this reason, the storage system will have 5 batteries. Wertz and Larson also uses a transfer efficiency of 0.9, as he presented in Table 11–40 (1999, p. 422).

With this information, the battery capacity can be calculated using the equation presented above:

$$C_r = \frac{P_t T_t}{(DOD)Nn} = \frac{(1x10^9) \times .08333}{0.50 \times 5 \times 0.9} = 3.073 \times 10^7 = 37 \text{ MW-hr}$$

To develop a complete picture of this system, the weight of the batteries must also be calculated. Wertz and Larson (1999) presents necessary data in Table 11–39 – Characteristics of Selected Secondary Batteries. This information includes the specific energy density (in W-hr/kg) and the current status. The table is re-created below, along with the calculated required battery weight based on the specific energy density and the required battery capacity:

Secondary Battery Couple	Specific Energy Density (W-hr/kg)	Status	Calculated Weight
Nickel-Cadmium	25–30	Space-qualified, extensive database	1.23x10^6 kg
Nickel-Hydrogen (individual pressure vessel design)	35–43	Space-qualified, good database	8.61x10^5 kg
Nickel-Hydrogen (common pressure vessel design)	40–56	Space-qualified for GEO and planetary	6.61x10^5 kg
Nickel-Hydrogen (single pressure vessel design)	43–57	Space-qualified	6.49x10^5 kg
Lithium-Ion (LiSO2, LiCF, LiSOCI2)	70–110	Under development	3.36x10^5 kg
Sodium-Sulfur	140–210	Under development	1.76x10^5 kg

Table 24. Battery Weight Calculations

Based on this information, the battery weight alone will be a significant contribution to the launch weight, which will in turn drive the cost-to-orbit up and decrease the overall cost performance. There is significant room to improve the battery capacity per kilogram, specifically as it applies to a space based solar power system. There may be potential to use a fuel cell or cells to store the energy in a more weight-effective manner. Driving the capacity-per-kilogram up and bringing the overall storage system weight down will only help the bottom line in the long run, even if the technology costs more to mature and integrate into the system initially.

b. Energy Transfer System

Throughout the research for this thesis, there are really two technologies that have the potential to be the basis for an energy transfer system for this type of satellite: a microwave system and a laser system. Based on current technologies, the microwave system is more well-known and established, giving it the highest technology readiness level (TRL) relative to the laser system. The microwave system also requires much larger structures, which presents its own problems with this type of satellite. The laser system, on the other hand, does not have the TRL to fly on a satellite today. The largest problem with the technology as it exists today is in the power generation, especially when it comes to generating a laser beam with enough power to transfer one gigawatt of energy. That being said, the laser system has a much higher potential for near-future operations than the microwave system. These two individual systems will be explored in the following subsection.

These opinions are codified in the 2007 NSSO report in Appendix A.

> If in space, the long distances and relative movement may require high frequencies in the visible or infrared range. If transmitting from space to Earth, the transmittance or opacity of the atmosphere must be taken into consideration. Generally speaking, there are only a few desirable windows of transmission where most of the energy of the beam is not scattered and absorbed. These include the visible, infrared, and lower radio frequency ranges. Visible and infrared ranges, because of their much shorter wavelength, have the advantage of much smaller apertures, but today have lower efficiencies both in generation and reception, are less mature at high power levels, may have eye-safety concerns, and may be unacceptable to

the public regardless of the density of the beam because of negative associations Light Amplification by Simulated Emission of Radiation (LASER) may have with the general public. Nevertheless, the ability to achieve high power at much lower weights, and the ability to transmit it to much smaller receivers deserves additional study and attention. The more typical design for Space Power Satellites has been the 2.4 or 5.8 GHz ranges where transmission and coupling is favorable. The disadvantages of this approach is the unforgiving physics of microwave power transmission, which requires extremely large apertures, and therefore large on-orbit weights, to mitigate the beam divergence. This minimum aperture to ensure a sufficiently small spot size and coupling efficiency is true regardless of the amount of power transmitted, and therefore scales poorly for small amounts of power. (NSSO, 2007, p. A-1)

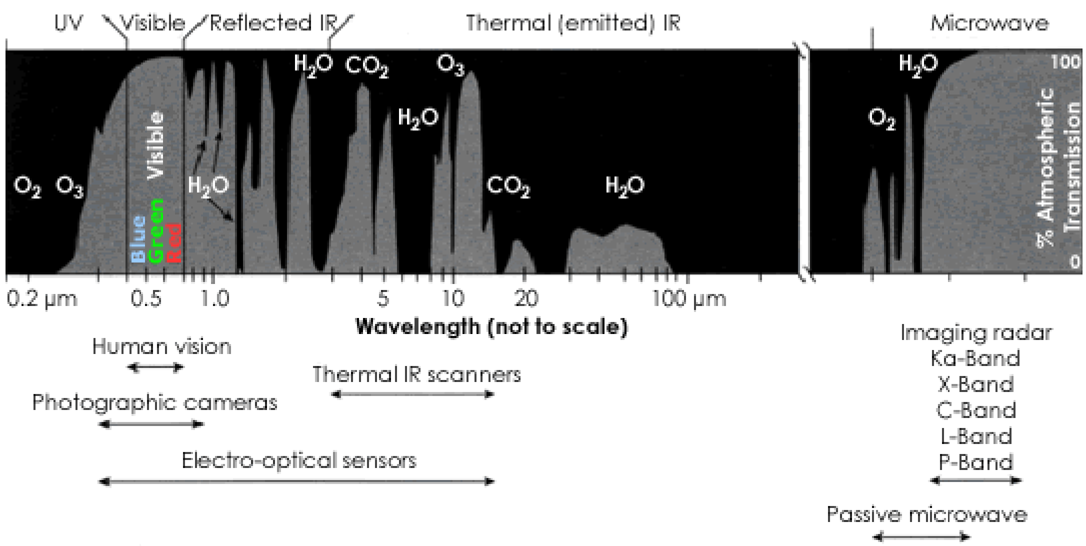


Figure 20. Transmission Rates of the Atmosphere (From King & Herring, 2002)

The first important thing to consider when choosing between these two different technologies is their ability to transmit power within their specific portions of the electromagnetic spectrum. Transmission through space is not the concern, but rather through the atmosphere. Due to the chemical make-up of Earth’s atmosphere, there are specific portions of the electromagnetic spectrum that do not pass through the atmosphere. If either the laser or microwave systems transmit in those portions, it would be ineffective as a power transfer system. Fortunately, based on the information presented in Figure 20, both the visible wavelengths (for the laser system) and the microwave

wavelengths have high transmission percentages. Therefore, the consideration for atmospheric transmission is identical between the two technologies and does not impact the final design of the spacecraft.

The microwave transmission system has been a central piece of the space based solar power discussion since it was initially discussed in the 1970s. At its most basic level, a microwave transmission system is quite simple, consisting of a transformer, the emitter, a rectenna (rectifying antenna) on the ground, and another transformer. The system efficiencies are very high for this type of system, making it a very attractive technology. It is also very mature, meaning that design and integration into the system would be relatively simple and straightforward. This system also has limitations, mostly in the weight and size category. Due to the physics of transmission, the emitter has to be sufficiently large to maintain a coherent beam. According to the NSSO report, this means the emitter on the spacecraft needs to be one kilometer in diameter (2007, p. 23). An antenna that large is going to have a significant impact on the overall cost and, therefore, the dollar-per-kilowatt metric that is so important for this system.

The other possibility for the power transmission payload is a LASER-based system. To date, the largest problem facing the laser system was a relatively low technology readiness level for a system that would provide added benefit to a space based solar power system. In previous iterations of the space solar power architecture, a laser system was too large, too heavy, and not nearly mature enough to plan around. The microwave system made much more sense. Lasers, on the other hand, have undergone significant technology advances in the last 30–40 years, enabling it's inclusion in this discussion. "Since the advent of lasers over four decades ago, solid state and gas lasers have followed largely separate development paths, with gas lasers being based either on direct electrical discharge for pumping or luminescent chemical reactions, and dielectric solid-state lasers being pumped by flash lamps and semiconductor diode laser arrays. [The Lawrence Livermore National Laboratory's] diode pumped laser is a new class of laser system that has been under development at LLNL for the past several years" (Rubenchik et al., 2009, p. 8).

To be sure, this does not necessarily imply a fully operational system prepared for spaceflight. There is more development and engineering needed to perfect the system. However, based on the remainder of the system, there is time for some development of the laser payload prior to system deployment. The LLNL diode pumped laser shows significant promise based on current experiment models. “Based on experimentally validated first-principles physical models, we predict that power-scaled systems will achieve unprecedented optical-to-optical efficiencies of 65–70 percent using today’s diode arrays, and enable fully packaged systems at <5 kg/kW (system mass to power output). This value is consistent with another laser under development, the High Energy Laser Area Defense System (HELLADS) being developed by the Department of Defense” (Rubenchik et al., 2009, p. 8). “Using high-efficiency (65 percent) diode laser pump arrays, overall laser system efficiencies (wall-plug) of ~50 percent are possible” (Rubenchik et al., 2009, p. 9).

The same paper provides insight into the weight necessary for this diode pumped laser system as well. Rubenchik et al., states that the diode pumped laser will “enable fully packaged systems at <5 kg/kW (system mass to power output)” (2009, p. 9). This means that at its heaviest, a 1 GW laser system should weigh in at approximately 5,000,000 kg. It is also likely based on Rubenchik’s description that it will be less than that. However, for planning and design purposes, the laser system mass will weigh in at 5,000,000 kg.

Both payloads have advantages and disadvantages, and it is critically important to ensure the trade-off between the two maximizes the return on investment and minimizes the overall cost of the program. Based on all the research presented above, it is the recommendation of this thesis to employ the laser transmission system on the space solar power satellite being designed. The overall size and weight savings outweigh the relatively low technology readiness level. As stated in the assumptions, it is going to take an international coalition and significant population education in order to properly operate the system, since there will be a natural aversion to a high power laser being operated from space and pointed at the general population.

E. SPACECRAFT SIZE AND WEIGHT

With the spacecraft bus and other systems designed or chosen, the next logical step is to combine these individual pieces and look at the system holistically. This system-wide design check will enable an informed discussion on the available launch vehicles and, therefore, an initial look at the cost-to-orbit. The pieces of this system that will factor into the initial weight estimate are the spacecraft bus, the solar panel, the storage system and the payload. Table 25 lists the segments of the solar power system and their corresponding weights as designed in the previous subsections. With that information, the last cell of the table represents the total weight of the system, which will be needed in the next subsection to examine the options for launch vehicles. This does not include the support system necessary for the solar panel, which is not provided by the selected satellite bus.

System	Weight
Boeing 702HP Bus	5,400 kg
Solar Panel	100,000,000 kg
Power Storage	176,000 kg
Diode-pumped Laser Transmitter	5,000,000 kg
TOTAL SYSTEM WEIGHT	105,181,400 kg

Table 25. System Design Weights

1. Potential Launch Vehicles

As examined earlier in this thesis, there are multiple launch vehicles seeking to serve a variety of customers. In the case of this space-based solar power system, a heavy-lift vehicle is an absolute necessity. For this reason, the launch vehicle discussion will focus on the Delta IV-Heavy and the Falcon 9-Heavy. These vehicles have the highest weight capabilities to a geosynchronous orbit where the satellite will eventually be

deployed. These launch vehicles, however, are constrained not only by their weight capability but also by the payload fairing size. For the purposes of this thesis and the demonstration of the sizes and weights involved, it is assumed that a portion of the space based solar power system that reaches the maximum weight of the lift vehicle also completely fills the payload fairing. While this may not produce a highly accurate cost estimate, it will be within the right order of magnitude.

The NSSO examined the launch vehicle market as it existed in 2007, before the emergence of SpaceX and the Falcon 9 Heavy launch vehicle. It is therefore not included in the discussion of the NSSO when referring to heavy-lift launch vehicles. In the report it states that "the SBSP Study Group found that the nation's existing EELV-based space logistics infrastructure could not handle the volume or reach the necessary cost efficiencies to support a cost-effective SBSP system. America's existing space manufacturing base is not suitably aligned at present for full-scale SBSP deployment" (NSSO, 2007, p. 32). The Falcon 9 Heavy vehicle is cheaper than an EELV in the same weight class. While there may be some overhead cost decrease, there is no appreciable difference between the Delta IV Heavy and the Falcon 9 Heavy. This is a troubling topic as it means the entire space manufacturing and launch markets would have to undergo a revolution in manufacturing and operations for a space based solar power satellite to be cost effective enough to launch.

2. Cost-to-Orbit

The final calculations necessary in this section revolve around the number of launch vehicles it will take to launch a space based solar power satellite into orbit, their associated costs and what the cost-per-kilogram is to launch the entire satellite into orbit. Again, it is assumed for the purposes of this thesis, that if the launch vehicle reaches its maximum liftoff weight, the payload fairing is also full. This precludes any potential issue on how to properly divide the satellite to fit within the payload fairing.

Based on the proposed orbit as well as the sheer size and weight of the satellite, it is expected that the launch costs will be extremely high. The 2007 NSSO report also pointed this out. It states that "today the United States initiates less than 15 launches per

year. Construction of a single SBSP satellite alone would require in excess of 120 such launches. That may seem like an astounding operations tempo until one considers the volume of other transportation infrastructure. For instance, in 2005, Atlanta International Airport saw 980,197 takeoffs & landings alone, an average of 1,342 takeoffs/day, or about 1 every minute 24 hours a day" (p. 31). One of the NSSO findings states that "the SBSP Study Group universally acknowledged that a necessary pre-requisite for the technical and economic viability of SBSP was inexpensive and reliable access to orbit. However, participants were strongly divided on whether to recommend an immediate, all-out attack on this problem or not" (2007, p. 31). The associated recommendation from the study group was "that NSSO, NASA, DOC, and other U.S. Government agencies should engage with industry (aerospace, energy, space tourism & manufacturing) to determine industry's level of desired industry/government cooperation for creating SBSP-enabling spacelift and supporting in-space transportation and logistics infrastructure" (NSSO, 2007, p. 32).

	Delta-IV Heavy	Falcon 9 Heavy
Satellite Mass	105,181,400 kg	105,181,400 kg
Vehicle Capability to GTO	13,399 kg	12,000 kg
Number of Vehicles	7850	8766
Cost Per Vehicle	$140,000,000	$128,000,000
Total Cost for all vehicles	$1,099,000,000,000	1,122,048,000,000
Cost Per Kilogram	$10,448/kg	$10,667/kg

Table 26. Launch Calculations

Table 26 demonstrates the high number of launches and the astronomical cost associated with launching a single space based solar power satellite as designed in this thesis. The NSSO report examined a concentrator-type system which significantly

decreased the number of launches because the solar array was significantly decreased. However, there is not much information publicly available on these systems to properly understand and codify them for inclusion in this thesis. This explains the large difference between the 120 launches described in the NSSO report and the approximately 8000 launches necessary according to this thesis.

The capability physically exists to launch the space based solar power satellite as designed in this thesis. It will take 7850 launches to do so, but it can be done. A key finding of this thesis is that the NSSO report on the launch market still holds true today, even with the introduction of SpaceX and the Falcon 9 Heavy into the market. It does not appear as though SpaceX has made enough of a cost savings to turn this endeavor into a worthwhile exercise. Another issue is that these vehicles do not move enough weight to a geosynchronous transfer orbit (GTO). While decreasing the cost of an individual launch is important, it will also be critically important to increase the weight-per-launch to GTO moving forward. Clearly, the combination of more weight to orbit for decreased cost will help enable the space based solar power system. In the short term, this is unlikely to happen.

F. CHAPTER SUMMARY

This chapter took the orbit decision from the previous chapter and designed the space based solar power satellite to operate in that environment. The specific spacecraft systems or subsystems that were explored and designed within this chapter was the payload solar array, the production satellite bus, the power storage subsystem and the power transfer payload. Once the satellite was designed and the weight calculated, the cost-per-kilogram to orbit was calculated, demonstrating that the current launch market is unable to support the high operations tempo necessary to launch a space based solar power satellite as also found in the 2007 NSSO report.

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V. APPLICATION OF STUDY

A. INTRODUCTION

To this point, this thesis has laid out the assumptions, requirements, and design elements of a space based solar power satellite system. It focused specifically on the space segment of the system in order to focus the discussion on the most difficult portion of the system to design and build.

B. CORRELATING RESEARCH

There has been significant correlating and related research, especially since 2005 in light of the ever-expanding fossil fuel use and a need to concentrate on moving toward renewable energy sources. The government as well as industry have been looking into the space based solar power construct and determining what the necessary critical path is for the system to be affordable as well as technologically feasible. As the technology has matured over the last decade, a space based solar power system has become more feasible.

As demonstrated throughout this thesis, one of the driving reports over the last decade was the National Security Space Office (NSSO) report on space based solar power from 2007. In it, the report discussed many of the issues raised in this thesis, especially as it pertains to the ground segment, the launch costs of the system, the legal challenges with such a system, and the overall business model feasibility of developing and operating such a system. While the report did not go into significant detail about the specific design chosen by the panel to explore, the report provided an excellent single point of reference for many of the common issues and solutions encountered when working with space based solar power. In the end, the report came to the conclusion that, while technologically feasible at the time the report was written, it was not feasible from a business model view nor was it feasible from an operational program view. There were too many roadblocks and difficulties with all of the individual parts, and not much has changed in the intervening five years since that report was written.

Another piece of related research was the report written by Lawrence Livermore National Laboratory (LLNL). It was a report detailing the design of a technical demonstration spacecraft with a space based solar power collector and transmitter on board. It was written by LLNL specifically because of the laser-based energy transfer system which is based on a diode-pumped laser under development at LLNL. It is that laser system that formed the basis for the system presented in this thesis. That was one of the first times a laser-based transfer payload was presented as a legitimate option for a space based solar power system. Until that payload was presented in that paper, and often since, the space based solar power architectures focused on a microwave transmitter which has a higher technology readiness level than the laser payload. While that is an accurate assessment of the technology readiness, the authors of the LLNL paper believe that the investment into the diode-pumped laser payload will pay huge dividends in the end due to a significant weight savings on-board the spacecraft.

Based on the increase in fossil fuel costs, the business model for space based solar power has also started to generate significant interest in the commercial realm. There has been a significant increase in the number of companies looking at space based solar power as not only a part of their portfolio, but in fact as the sole focus of the company. There are three specifically that have made significant strides in the space based solar power arena: Solaren, Space Energy and PowerSat Corporation. While Solaren and Space Energy are not providing any proprietary information that could expose them in the long-term to increased competition and risk, PowerSat has published a white paper outlining their notional first satellite. It appears to operate much like the satellite outlined in this thesis with one important distinction: the solar array. "The Independent Solar Energy Converter (ISEC) is launched into low earth orbit as a single unit, then inflates its outer support surface, deploying the solar array. Electricity from the array is used to power ion (or Hall-effect) thrusters, enabling the ISEC to transport itself from low earth orbit to geostationary orbit without the use of a chemically powered space tug. The modular design eliminates the need for on-orbit personnel; each ISEC can dock with units already in place and begin functioning without astronaut assembly. The ISEC concept is unique to the PSU-1 design, and is intellectual property of PowerSat Corporation" (Maness &

Hendrickson, 2010, p. 3-4). This is a very unique attempt at decreasing launch costs—by turning the large solar array into small independent pieces that then re-assemble once in orbit to aggregate to form a larger array. This method could have a definite impact on the bottom line metric—the dollar-per-kilowatt and will require additional investigation.

C. RECOMMENDATIONS

The space based solar power system as designed through the course of this thesis requires monstrous solar array areas, enormous launch costs, and an unimaginable spacelift campaign. Despite the fact that some of these design points appear unfeasible, there are some recommendations that can be made to make progress toward economically and operationally feasible space based solar power designs.

1. Operational Orbit

As discussed in an earlier chapter, there are four main orbital regimes for satellites to operate in: low earth orbit, medium earth orbit, highly elliptical orbit, and geosynchronous orbit. Each one of these orbits provides different advantages and disadvantages based on the mission being performed by the satellite and the desired outcomes. Additionally, there may be more than one correct design choice depending on the mission.

A low earth orbit has the distinct advantage of being the closest to the launch pad, so a single launch vehicle can launch significantly more weight into a low earth orbit than any other orbit. It falls behind, however, when single target access time is compared to the other potential orbits. Low earth orbiting spacecraft have up to fifteen minutes of contact with any one site on the ground during a pass. It also falls behind when looking at the revisit opportunities, which are extremely limited with low earth orbiting spacecraft.

The middle earth orbit offers increased access to a single point during the course of one orbit. It also offers a compromise in launch cost between a low earth orbit and a geosynchronous orbit. It does not, however, solve the problem a low earth orbiter has of revisit opportunity. The trade, however, is the increased access time during a single orbit, decreasing the necessity for a high revisit rate.

A highly elliptical orbit has the potential to solve the access duration and revisit frequency, but comes with its own issues. The highly elliptical orbit was designed to provide a long dwell capability over a specific hemisphere, primarily for communications satellites. This significantly increases the access time over one specific hemisphere, but minimizes access to the opposing hemisphere. This orbit would effectively limit all space based solar power capability to the northern hemisphere. The biggest downside to a highly elliptical orbit is that perigee of the orbit is very low, meaning that the satellite will encounter significant atmospheric drag effects on every revolution. A massive satellite like the space based solar power satellite would encounter heavy stresses on the entire satellite and would also require constant burns to maintain its orbit.

The last orbital regime available is the geosynchronous orbit. This option comes at the huge price of cost-to-orbit. Due to the distances involved, a launch vehicle cannot carry nearly the same spacecraft weight as it could into a low earth orbit. However, the geosynchronous, or more accurately, geostationary, orbit comes with a huge upside. It has access to almost half of the Earth at one time, maintains that access twenty-four hours a day, and has near-constant solar illumination. The constant illumination and constant access make up for the high launch costs because the ability to constantly collect solar energy and transfer it to earth drives down the dollar-per-kilowatt figure that is critical for the business model of a space based solar power satellite.

It is the recommendation of this thesis that, for an operationally deployed space based solar power system, the satellite be deployed into a geostationary orbit. While the transfer capability will be limited to slightly less than half of the Earth at one time, the near-constant solar illumination is critical to the success of a space based solar power satellite. The launch capability to deploy the satellite will be an issue, at least in the near-term, but as stated above, the increased collection and transfer capability enabled by the near-constant solar illumination overcomes increased launch costs.

2. Power Transfer Payload

The other significant design decision made during the course of this thesis was which power transfer payload would be selected for the satellite. Based on the

transmission windows of Earth’s atmosphere, the best options for a power transfer payload are transmitters that operate in the visible portion or the microwave portion of the electromagnetic spectrum. There are important factors to consider when making a design decision between these two systems, namely the technology readiness level, the size and mass of the system, and the efficiency of the system to convert power into the transmission medium and back again.

The microwave transmission system was the basis for the original space based solar power satellite designs. The technology and capability has existed since the 1970s and therefore has a high technology readiness level than the laser system, which is relatively new. The high TRL on this type of system makes it an attractive candidate for a space based solar power system, especially one that is attempting to design, launch, and operate in the near future. Little design work is needed in order to integrate a microwave system into a new system. Another significant advantage of a microwave system is the high conversion rate of the electricity, both from direct current (DC) power to microwave, as well as from microwave back to DC. This high conversion rate again makes this technology attractive.

A microwave system has a significant downside, however. Due to the nature of the transmission medium, the transmission antenna has to be large to account for the spread of the beam. A microwave transmission beam with the appropriate power output and power density over the distance from a geosynchronous orbit will require an antenna with a 500-meter-plus diameter. Some estimates place the size of the antenna at or larger than a kilometer in diameter. To date, the largest antenna placed in orbit is the 22-meter SkyTerra antenna (Boeing Corporation, 2010b, p. 3). Launching, deploying and operating such a large antenna is going to be a leap in technology and operations and places a large burden on the launch segment, since a 500-meter-plus antenna will not get to the geosynchronous belt without a significant launch campaign. As a corollary to this, the ground receiver-antenna (rectenna) is also a potential issue. The majority of microwave-based designs put the diameter of the rectenna around ten kilometers, including the 2007

NSSO report. While the phenomenology of microwave power transmission may make this antenna relatively transparent to the environment, it is still an enormous undertaking to build

By contrast, laser-based transmission systems have really only become part of the space based solar power discussion since the 1990s. Lasers are well understood, but their ability to transfer energy has only been explored recently. This makes a laser system a relative risk, especially when compared to a microwave system, for a space based solar power system. However, as demonstrated by the paper produced by the Lawrence Livermore National Laboratory, significant progress is being made to reduce the size of the equipment needed while increasing the power output. In addition, lasers by their very nature have a high energy which causes a host of potential problems. The first is the potential for a conjunction with another spacecraft. Since the space based solar power satellite is operating in a geosynchronous orbit, there is significant potential for a conjunction as described in an earlier chapter. While there are ways to mitigate this, it may mean pausing output while a satellite traverses the area under the beam. Another issue with laser energy is that there is a natural aversion to high power lasers transmitting in a populated area. The ground sites will have to be located appropriately to avoid any potential issues with the local population. Finally, lasers currently do not perform as well in the conversion arena as a microwave system does. Converting electricity to light and back again causes a significant degradation in the power transmitted across an area.

A laser has significant upside as well. As opposed to a 500-meter-plus antenna, a laser systems' aperture is much smaller. There is more equipment to be housed on the spacecraft bus, but overall there is a reduction in the volume footprint. There is also a corresponding decrease in the size of the ground footprint. This size decrease across the board is one of the things that makes a laser system an attractive option for a space based solar power satellite. Also, it appears as though, while the microwave systems have found their optimal operations design, there is much more experimentation and development potential for lasers, meaning that many of the current issues may no longer be an issue in the near future.

With all of these considerations in mind, this thesis recommends the continued investment into the technology to allow for transmission of energy via lasers. As demonstrated in the section above, there is enough time to develop the best possible laser system while waiting for cheap, rapid, and reliable space launch capabilities. Rather than building around an existing technology that may have reached its design limits, investing in lasers would be the best use of time and money. With enough time and money, significant strides can be made to increase the efficiency while decreasing the footprint (both volume and weight) to make it the optimum payload for a space based solar power system.

D. CHAPTER SUMMARY

This chapter focused specifically on correlating research as well as the recommendations of the thesis on specific design elements, namely the launch vehicle and the power transfer payload. There is significant research occurring as of the writing of this thesis whether directly or indirectly related to space based solar power satellites. Specifically, there are three public companies that are exploring space based solar power satellites as their specific business model. One of them, PowerSat, developed a similar design as this thesis, albeit with a significantly smaller power output. This significantly decreased their overall design cost, their weight and therefore their launch cost. In addition, their solar array is segmented so it can launch in individual segments and combine once on orbit, meaning PowerSat can design these segments to fit specifically within a payload fairing, making each launch as economic as possible.

There was also discussion on how to launch all of the pieces and parts of a space based solar power satellite into orbit. The sheer weight of the satellite makes the possibility of an operational satellite as designed in this thesis essentially non-existent. As designed in this thesis, it would take almost 8,000 launches of either the SpaceX Falcon Heavy or Boeing Delta IV Heavy to launch everything into its geosynchronous orbit. These two launch vehicles are the two heaviest lifters available on the market as of the

writing of this thesis. Without cheap, rapid, and reliable access to space, a space based solar power satellite will never be truly feasible regardless of the spacecraft technology available.

There is also significant research being done on power transmission via lasers. As stated in the previous section, there are more efficiencies that have yet to be realized with a laser system. It is important over the next decade to continue to explore new laser power transmission designs and find a design that maximizes not only the overall output but the system efficiency while minimizing the weight. When a laser energy transmission system reaches that level of design maturity, it will be ready for inclusion into an operational space based solar power system.

VI. CONCLUSIONS

A. KEY POINTS AND CONCLUSIONS

Regardless of current technical feasibility, profitability, or practicality, it is important to acknowledge that the potential for space based solar power to solve many of the nation's electricity needs. The increase of over 1100 W/m2 in solar energy available to a solar panel is well worth the investment into the technologies necessary to make space based solar power a reality. It is, however, going to take a significant investment over the next few decades to make space based solar power both profitable and practical. There are many hurdles to a practical and profitable space based solar power system including solar panel size and weight, relatively immature energy transfer technology, and astronomical launch costs. These hurdles can be overcome, however, through long-term research and investment. There are two ways to fund this type of long-term investment: by government research, such as at being done by Lawrence Livermore National Laboratory, and through commercial investment, such as that being done by Solaren, Space Energy, and PowerSat. It is encouraging to see the combination of research and investment being funded across the interested parties and as the interest in space based solar power grows, the mutual benefit will pay huge dividends.

This thesis set out to answer three specific research questions: what is the optimal orbit for a space based solar power satellite; what power transfer payload is appropriate for a space based solar power satellite; and is a space based solar power satellite economically feasible and advantageous based on the design presented in this thesis. Many different segments of this space based solar power satellite were presented during the course of this thesis, the majority of them related directly to the design and function of the satellite itself, as the other design requirements and other various factors were out of the scope of this thesis. A summary of the first two research questions appears below followed by an in-depth examination of the final research question.

The first question posed by this thesis was that of the operational orbit of a space based solar power satellite. The four traditional orbits were presented at length,

examining the pros and cons of each orbit and comparing those factors to the ultimate requirement of getting energy to the ground reliably and in a cost-effective manner. The latter factor had the greatest influence on the final operational orbit recommended by this thesis; that is, the longer the satellite is in view of both the sun and a ground site, the more power it can transfer and therefore the more cost-effective the system is. For this reason, a geostationary orbit is recommended by this thesis. This decision is supported by the commercial venture PowerSat, who has chosen the same operational orbit for their notional space based solar power satellite. However, due to the prohibitive costs, a pathfinder or experimental system would be much better suited to a low earth orbit. Although it will have limited access to the sun and a ground site at the same time, that orbit will minimize the number of launches required to put the satellite into orbit while still allowing a pathfinder satellite to prove technologies required to operationalize the system.

The second research question was to determine the appropriate power transfer payload technology for a space based solar power satellite. There are only two technologies currently in the running as the power transfer payload: a microwave system or a laser system. The microwave system has a much higher relative technology readiness level and currently has the highest conversion efficiency, making it an attractive and well-known technology. However, because of the phenomenology of microwave power transmission, the emitting antenna and the receiving rectenna are both prohibitively large. A laser transmission system, however, would have a much smaller emitter and a very small ground footprint compared to a microwave system. A laser system does not have the conversion efficiency or atmospheric transmission efficiency of a microwave system, but based on information from the Lawrence Livermore National Laboratory, a laser system has much room to grow over the next decade if the investment is made into the technology. For this reason, this thesis recommends investing in a laser transmitter for a space based solar power architecture.

The final research question initially posed by this thesis was the economic viability of a space based solar power system. This question appears very straight-forward on the surface. However, this can be a very complicated issue. The NSSO Study

Group realized this as well, in its report, “the the SBSP Study Group found that Space Based Solar Power is a complex engineering challenge, but requires no fundamental scientific breakthroughs or new physics to become a reality” (2007, p. 20). However, it goes on to state that “while the study group believes the case for technical feasibility is very strong, this does not automatically imply economic viability and affordability – this requires even more stringent technical requirements” (NSSO, 2007, p. 20). At the heart of the complexity is the intended distribution of the power being transmitted from space. Depending on its distribution, it may have different cost competition with fossil fuels. For example, the 2007 NSSO report gives traditional domestic energy costs of approximately \$0.04 per kilowatt-hour (p. 34). However, the same report found that overseas military installations are paying as much as \$1 per kilowatt-hour (p. 34). Therefore, if the space based solar power system served primarily overseas military installations, it would be considered cost-effective under \$1 per kilowatt-hour. The true test for a space based solar power system will be the competition with fossil fuels in the traditional domestic market. If a system can be competitive at cents-per-kilowatt-hour, it will truly be a game changing system.

There are important features of the space based solar power system that directly contribute to the overall cost of the system. The biggest contributor to overall cost is weight, influenced heavily by the energy transmitter and the solar array. The weight of the system drives the launch costs which, for a high energy system as envisioned by the NSSO report, places launch costs alone in the trillions of dollars. It is possible to estimate the total weight of the system (as was done in a previous section of this thesis), the total cost of the theoretical first unit satellite, including research and development costs, and therefore not only a comprehensive view of the total budget required for such a system, but also, over a given range of power output, a theoretical maximized power output with a minimum cost. This is done with the equations found in chapter 20 of Wertz and Larson’s book in Table 20-8. Given these equations and the largest drivers of system weight, the calculations, weights and costs appear below in Table 27.

Power Output on Ground (in kilowatts)	Total weight	Launch cost	Spacecraft bus cost	Solar array cost	Spacecraft cost	Payload cost	Total Cost	$/kg	$/kW
1000	110576.3598	$1,155,361,622	$1,328,753	$16,467	$1,345,221	$53,809	$1,157,426,535	$10,467	$1,157,427
50000	5264217.989	$55,003,397,158	$1,328,753	$22,407	$1,351,160	$54,046	$55,005,471,189	$10,449	$1,100,109
100000	10523035.98	$109,950,372,195	$1,328,753	$23,476	$1,352,229	$54,089	$109,952,447,866	$10,449	$1,099,524
200000	21040671.96	$219,844,322,268	$1,328,753	$24,549	$1,353,302	$54,132	$219,846,399,588	$10,449	$1,099,232
400000	42075943.92	$439,632,222,416	$1,328,753	$25,627	$1,354,381	$54,175	$439,634,301,390	$10,449	$1,099,086
800000	84146487.83	$879,208,022,710	$1,328,753	$26,711	$1,355,464	$54,219	$879,210,103,348	$10,449	$1,099,013
1000000	105181759.8	$1,098,995,922,858	$1,328,753	$27,060	$1,355,814	$54,233	$1,098,998,004,032	$10,449	$1,098,998
10000000	1051768998	$10,989,451,429,488	$1,328,753	$30,700	$1,359,454	$54,378	$10,989,453,516,249	$10,449	$1,098,945

Table 27. System Cost Analysis

With the known system weights and costs known, it is possible to view these in graphical form. The first graph, Figure 21, shows the total cost as a function of the output power, displayed on a logarithmic scale. The second graph, Figure 22, shows the dollar-per-kilowatt as a function of output power.

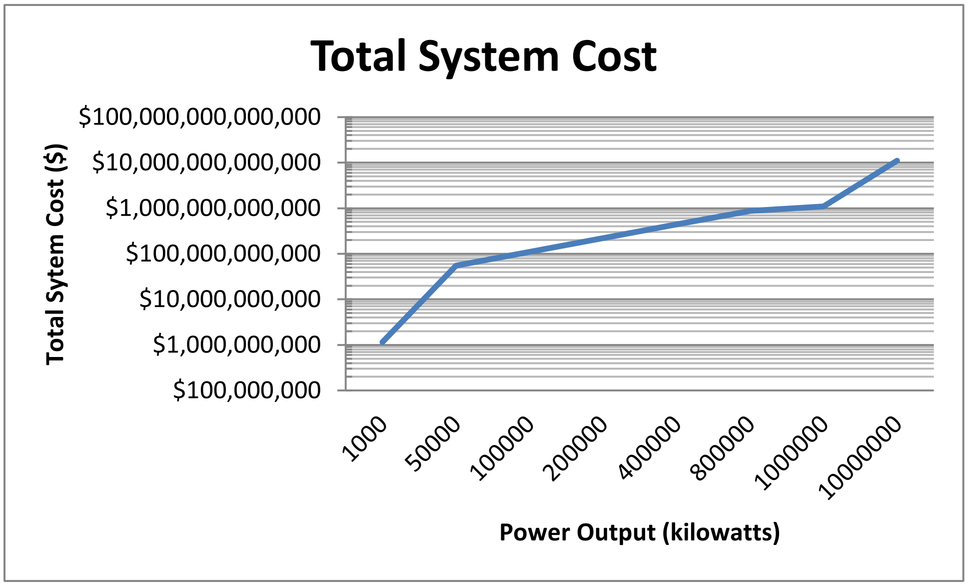

Figure 21. Total System Cost vs Output Power

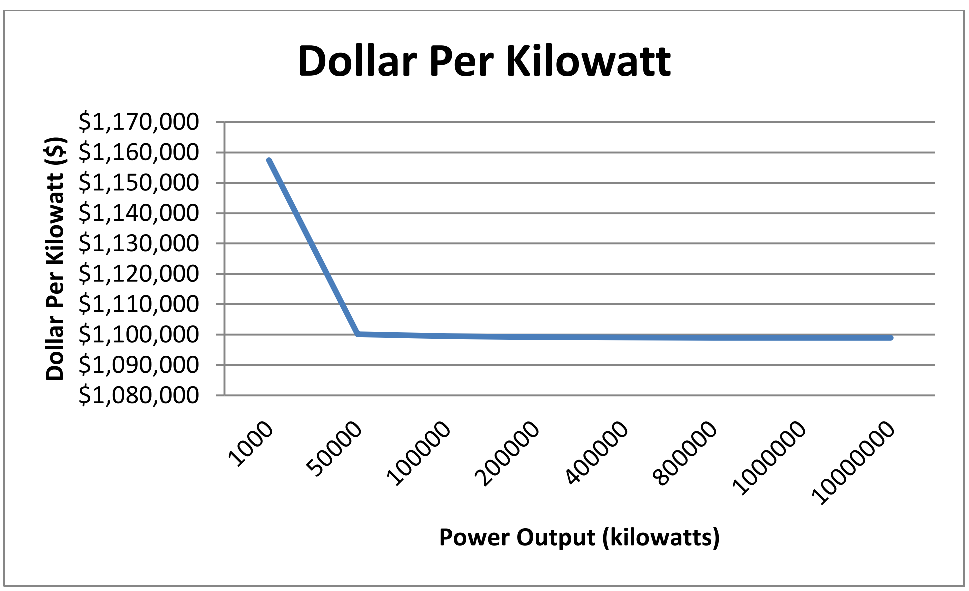

Figure 22. Dollar-Per-Kilowatt vs Output Power

These graphs show that there is little appreciable increase in overall system cost until the output power reaches over 1 GW. They also show that there is little decrease in the dollar-per-kilowatt statistic after 50 MW. Therefore, especially for experimental versions of a space based solar power system, it would appear the target output would be 50–100 MW.

The ultimate question, however, is whether or not a space based solar power system is currently economically and technologically feasible. The answer, therefore, needs to be two-fold. First, it is a question of whether or not the global population currently has the technological capability to develop and launch a system as described in this thesis. The second question is whether or not it is currently economically feasible to do so. These two questions are heavily inter-related, especially as it pertains to construction and launch.

The first question of technological feasibility. Based solely on the design of the satellite from this thesis, the answer is no. Specifically, the laser transmission system is holding the design back. The system being developed at Lawrence Livermore National Laboratory does not have the technology readiness level to be launched as an operational payload. It may soon be an option with a much smaller transmission power, but it is not currently available. That being said, a microwave system is feasible with today's technology. It is well understood and the design has been consistently optimized over the last few decades. In that respect, a space based solar power satellite, regardless of likelihood of launch, is, in fact, feasible using today's technology. While feasible, it is highly unlikely to be launched due to the relatively low launch weight and high launch cost available on today's launch vehicles. Before such a system could be launched, cheaper access to space is a must.

Second is the question of economic feasibility. Based on current technology, it is highly unlikely that, in the current budget-constrained environment, any true movement towards a space based solar power system will occur. Based on the information presented in the previous section, the cost and power output for the space based solar power system designed in this thesis does not meet the cost requirement to compete with fossil fuel-based energy. It will take a significant up-front investment in the system to bring it to

reality, which is currently not feasible given the current economic climate. With some technology investment, however, some of the system cost could be spread out over the next decade (or two) by generating more efficient solar cells, by finding a more efficient means of power transmission, and by pushing for consistently cheaper, faster, and more reliable access to space. With some of those costs spread out now, it will be cheaper overall when the economic climate improves and a space based solar power system becomes a reality.

Given the current state of both technology and the global economy, a space based solar power system is not feasible. As described previously, there are multiple barriers to such a system including a technologically-immature power transfer payload and astronomic launch costs, partially due to the size of the required solar array. However, because a system is not currently feasible does not mean that research should halt and those costs transferred to the future. It is possible to solve many of the current system barriers with a small investment in technology and systems engineering now and allow the fruits of that research to save money in the long run when the economic climate would support such a large up-front investment. Space based solar power is a potential answer to the rapidly dwindling fossil fuel sources and it will provide a consistent, stable and renewable source of energy for a long time once the initial investment is made.

B. AREAS TO CONDUCT FURTHER RESEARCH

There are numerous areas within the space based solar power construct to conduct further research. Each of these areas may or may not bear any real benefits to the system, but throughout the course of this thesis, they have stood out as possibilities for reducing overall system weight (and therefore, cost) or increasing the overall system output, thereby reducing the required input power and reducing the overall system cost. These areas are overall system design, solar array technology, and power transmitting technology.

The first area, overall system design, was first brought up while examining the laser transmission system being developed by Lawrence Livermore National Laboratory and in the 2007 NSSO report. Both of these reports detailed a system based not on a

linear solar panel, as the satellite developed by this thesis was designed, but rather based on a system of reflectors that concentrate solar energy onto a significantly smaller solar array. This allows the overall incident solar energy to increase at the solar panel which allows for a decrease in the solar array size while maintaining the same system input power. While this type of system has been notionally designed, there is currently little information available, especially as it pertains to the design of the reflectors. To properly design and evaluate a reflector-type system, it would be important to know the weights and design cost for these and specifically how it relates to the size of the solar array. In an earlier chapter, it was stated that a concentrator-type system achieved a 300 percent increase in output power over a similarly-sized standard solar panel, but that is too generic and does not contain specifics on the relationship between the reflectors and the array. Prior to examining that system in-depth, the reflectors and their relationship to the system would have to be researched and understood.

The second topic to conduct further research on is the solar array. This is tied to the previous paragraph in that it is important to understand how specific types of solar arrays could perform in a space based solar power construct, especially as it relates to a concentrator-type system. It is also critical to continue to research the best theoretical and production solar cells to look for higher efficiencies which would allow the array size to drop. While it is highly unlikely to ever reach 100 percent efficiency, it is possible, and likely, that efficiencies will continue to increase over the next few decades. As these efficiencies increase, the calculations done within the construct of this thesis should be revisited with the newest applicable efficiencies and another feasibility study undertaken.

Finally, over the next few decades, it will be vitally important to continue to explore and research new ways to transmit energy from a spacecraft to Earth and, more specifically, how to maximize the transmission power while minimizing the system weight. The studies being done at Lawrence Livermore National Laboratory is already a step in that direction. Their design, while not completed nor available for launch soon, opens the door for further research and development into high-energy transfer systems. As these systems mature and increase in their technology readiness levels, they should be

included in a feasibility study immediately as they may change the overall cost-benefit equation and tip it in the direction of being economically feasible.

While there are other efficiencies that could be gained to increase the overall economic feasibility of a space based solar power system, these three specific topics could have a heavy influence on the overall feasibility of such a system. With continued research and development underway in both the government and private sectors, it is likely that a space based solar power system will become economically and technologically feasible as a means for collecting and distributing the shining example of "green" energy - the ever-renewable energy from our Sun.

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